

NASA-CP 1991 International  
3160 Conference on Aging  
Aircraft and Structural  
Airworthiness

*NASA Conference Publication 3160*

# **1991 International Conference on Aging Aircraft and Structural Airworthiness**

*Proceedings of an international conference held in  
Washington, D.C.  
November 19-21, 1991*



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*NASA Conference Publication 3160*

# **1991 International Conference on Aging Aircraft and Structural Airworthiness**

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*Hampton, Virginia*

Proceedings of an international conference sponsored by  
the Federal Aviation Administration and the National Aeronautics  
and Space Administration, Washington, D.C., and held in  
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**1992**



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## PREFACE

Civil aviation serves a strategic interest to the United States. Thus, the need to maintain a high standard of aviation safety is axiomatic. While the industry has witnessed phenomenal growth in the past two decades, the proportion of aging aircraft in air carrier fleets has been ever increasing—a trend that is projected to continue.

Certain events which occurred in 1988 and previous years and public concern about the safety of the high time aircraft led to an International Conference on Aging Aircraft in Washington, D.C., during which a number of aviation safety related issues applicable to aging aircraft were raised and discussed among the conference participants. Various groups that have a vital stake in commercial aviation made certain recommendations which led to the formation of a task force and a broad undertaking involving the Federal Aviation Administration (FAA), the National Aeronautics and Space Administration (NASA), industry and international organizations to insure continued airworthiness of aging airplanes. At the same time, significant research and development programs were initiated by the FAA and NASA to complement the broader effort.

In view of the success of the 1988 conference, the FAA desired that conferences be held annually to disseminate information on the status of the task force's activities on aircraft certification and airline maintenance related issues and to offer a forum for participation by all interested parties. Thus, the 2nd International Conference was held in 1989 in Baltimore and the third in the series in 1990 in Bordeaux, France. Both of them, like the first conference, attracted rousing attendance and interest. Thus, the stage was set for organizing the 4th International Conference in 1991.

By 1991, the FAA and NASA research programs were yielding significant results, a consideration that led to a larger segment of the 4th International Conference Program being devoted to presentations about the impact of the research on the other activities of the task group. Agreement had also been reached between the FAA and NASA to coordinate and integrate their respective program activities. Hence, NASA was co-opted as a sponsor of the conference.

A group consisting of Mr. Stephen Erickson of the Air Transport Association of America, Mr. Jesse Lewis and Mr. Richard Johnson of the FAA, and Mr. Thomas Crooker of NASA was formed to organize the conference. The members of the group are owed special thanks for their efforts that ensured the success of the conference.

Dr. Sam G. Sampath, FAA Technical Center  
Dr. Charles E. Harris, NASA Langley Research Center



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**Introductory Remarks**  
**Anthony J. Broderick**  
**Associate Administrator for Regulation and Certification**  
**Federal Aviation Administration**

The United States, together with the joint aviation authorities, has been heavily involved in the aging aircraft issue since the tragic Aloha accident in 1988. Together with the Air Transport Association of America and other industry groups, we have established a number of programs designed to address the continuing airworthiness of older aircraft.

For large transport aircraft, we have

- (1) Established structural modification programs eliminating continued inspections in critical areas which are difficult to inspect.
- (2) Conducted an assessment of the aging U.S. fleet.
- (3) Defined programs of directed inspection in corrosion prone areas of the aircraft structure.
- (4) Developed criteria for assessment of structural repairs to ensure their durability.
- (5) Developed maintenance program guidelines.

We are also reviewing the service difficulty problems encountered in the aging commuter fleets and identifying actions which can be taken to ensure the continued airworthiness of this fleet.

This week you will hear, in some detail, about these ongoing programs. But, I would like you to concentrate more on what we can do to address the airworthiness of aging aircraft. You will hear from a number of noteworthy authorities in various technical areas for the next three days. You will also have the opportunity to review ongoing research efforts sponsored by both NASA and the FAA. We ask that you apply your experience to help us implement and enhance what is being done. Take back with you things you see here that can be used. Ask us questions where we have not made it clear what we are doing or where we intend to go with these efforts. When you return home, write us and tell us about what your country is doing and where you find that our efforts can be merged to ensure the safety of the international aging aircraft fleet.

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REMARKS BY FAA ADMINISTRATOR  
JAMES B. BUSEY  
1991 INTERNATIONAL CONFERENCE  
ON  
AGING AIRCRAFT AND STRUCTURAL AIRWORTHINESS

On behalf of NASA Administrator Dick Truly and myself, I want to welcome you all to this international conference. I am delighted to see the large turnout and the outstanding lineup of speakers over the next three days.

I am particularly pleased that Congressman Tom Lewis will be speaking to the conference on Thursday. Congressman Oberstar was scheduled to be here today, but he couldn't make it. I am pleased, however, that Congresswoman Joan Kelly Horn, the Vice Chair of the House Aviation Subcommittee, is here in his stead.

I just want to take a minute to comment on Congressmen Oberstar and Lewis. Both of these gentlemen have been leaders on Capitol Hill on aviation issues, and aviation safety issues in particular.

Congressman Oberstar has been a true champion on the aging aircraft issue. As Chairman of the Aviation Subcommittee of the House Public Works Committee, he has been highly effective in using his leadership role to advance this cause as well as other important aviation issues.

In fact, he was the driving force behind the Aging Aircraft Safety Act of 1991. Essentially, this requires the FAA to initiate rulemaking action leading to a rule that would assure the continuing airworthiness of aging aircraft.

This is a significant step forward in promoting aging aircraft safety because it puts the cap on many of the programs we have already established and the initiatives that are currently underway.

So I think the aviation community and the traveling public owe Congressman Oberstar a debt of gratitude for his dedication in promoting this important cause.

I might just note that we have already put together a team to accomplish the objectives of the legislation. The team is now at work developing a proposed rule, and they are shooting to have a first draft ready for internal review by mid-December.

I also want to recognize the contributions of Congressman Tom Lewis. As the ranking member of the House Subcommittee on Technology and Competitiveness, Congressman Lewis has been very supportive of FAA programs, including aging aircraft initiatives.

In fact, in 1988, Congressman Lewis was very instrumental in passage of the Aviation Safety Research Act that led to the National Aging Aircraft Research Program.

He also has been very effective in promoting aviation safety legislation in other areas, particularly on cabin safety issues and fire safety research.

So, Congresswoman Horn, I hope that you would pass on to both the Congressmen our appreciation for what they have done. And, again, I am glad you could join us today.

As you all know, this conference is the fourth in a series of international forums on the aging aircraft issue that have been held annually since 1988.

I think it is truly phenomenal the way the international aviation community has come together to focus on this problem of aging aircraft since the Aloha accident in April 1988.

Although FAA and other civil airworthiness authorities had been working on the issue of aging aircraft for many years before that, the Aloha incident galvanized the entire aviation community like no other event in recent memory.

The Aloha incident was a wakeup call because it challenged our previously held assumptions about aircraft airworthiness and our philosophy behind the aircraft maintenance program.

As a result, it brought us all together in an unprecedented way to focus on what needed to be done to ensure the continued structural safety of the world's airline fleet.

Since the first international conference here in Washington in June of 1988, a great deal has been accomplished. And I would like to take this opportunity to commend everyone who has made this progress possible.



From the FAA, I want to thank Tony Broderick in particular, as well as Craig Beard and Tom McSweeny, for their leadership roles in this area.

I also want to especially recognize the key role that the Airworthiness Assurance Task Force has played. Shortly after the first conference, this group was formed as a result of a recommendation from ATA and AIA.

What it did was bring together a group of technical representatives from the airlines and aircraft manufacturing industry around the world to work on the issue.

This task force has been an extraordinarily valuable resource. The FAA has taken several important actions based on its recommendations, particularly in the area of structural modifications and corrosion control.

In fact, the group has been so valuable that it is now part of a regular FAA advisory committee. Currently, we are assessing recommendations from the Task Force on maintenance planning, fatigue design, and testing, and we are expecting additional recommendations in the near future.

So, we are grateful for their expertise and support.

Over the next three days, you will be hearing about many programs and plans, so I don't want to get into details. But, I do want to highlight some of the broad areas of accomplishments I think are worth noting:

- \* FAA has issued several new airworthiness directives calling for the modification or replacement of certain aircraft components on aging aircraft. Since March 1990, as a matter of fact, structural ADs have been issued for the majority of Boeing, Douglas and Lockheed aircraft.

We are currently working on additional ADs for a few remaining aircraft, as well as on ADs for dealing with problems of airframe corrosion. We expect that all structural and corrosion-related ADs will be completed and issued by January of next year.

- \* Moreover, we have established an aggressive four-year schedule for complying with these ADs. This schedule was adopted with industry input and public consultation with hundreds of individuals.

Contrary to earlier concerns expressed by GAO, we do not believe the 1994 deadline for compliance is unrealistic. In fact, I am confident that 1994 is an attainable and realistic goal.

\* I also am confident that we have in place an effective program for verifying compliance with the ADs. Already, our principal maintenance inspectors are working with airlines to gather compliance data.

By the end of the year, we expect to complete a data base on every aircraft operated by U.S. airlines. This will include a plan on how the operators will ensure that every aircraft complies with these ADs.

A good example of our commitment to monitoring aging aircraft is our National Work Program. Under this program, a team of experienced inspectors and engineers are conducting "hands-on" evaluations and examining the maintenance programs of the U.S. carriers involved.

The findings from these evaluations are being incorporated into our on-going National Safety Inspection Program.

\* We also have initiated training programs for our inspectors. The initial focus has been on corrosion, but the program is being expanded to include the assessment of fatigue and the durability of repairs.

\* In addition, we have broadened the scope of the aging aircraft program to include commuter aircraft. This initiative consists primarily of reviewing existing airworthiness directives and service bulletins relating to commuter aircraft and to establishing guidelines for supplemental structural inspection programs.

\* Finally, I would like to mention the National Aging Aircraft Research Program. This program is the result of the Aviation Safety Act of 1988 which gave us the enabling authority and the tools to launch an extensive research program. This program is aimed at ensuring the airworthiness of the existing fleet as well as future transport aircraft.

It is the largest program in our aircraft safety research area, and it is directed to a whole gamut of concerns--material fracture and fatigue, human factors, maintenance and inspections.

Significantly, this research program also incorporates a transfer of technology agreement with the Departments of Defense and Energy, as well as NASA.

Later, you will be hearing about the specific benefit we already have reaped from this program.

So, in short, I think you can see that we are attacking the problem from all angles, and I just want to say once again that I am pleased at the progress that we are making.

However, let me quickly add, pleased is one thing--satisfied is quite another. I don't think anyone in this room is satisfied that all that can possibly be done is being done.

And that's why we are here today--to discuss what we have learned, but more important to identify what still needs to be done to ensure the continued airworthiness of the world's air fleet.

And you won't get it all done this year either. Many of you will be back next year--if not here in Washington, then some place else--discussing many of these same issues. It's a job that is never finished.

It's is an awesome responsibility, but as I look around this room, I know it's in good hands.

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**1991 International Conference on Aging Aircraft  
and Structural Airworthiness - Background  
A. A. Hale, Vice President-Engineering  
American Airlines, Tulsa, Oklahoma**

I appreciate the opportunity to share a little bit of the background for the continued airworthiness programs that we have underway today. I think it's important to remind ourselves once in a while about the reasons we have such an extensive effort in place, and what we're getting out of it.

I can't think of any other accident that has caused the industry and public reaction that has stemmed from the Aloha Airlines incident in April 1988. There hasn't been a similar example of structural failure in the jet age. It merited the wholehearted attention of operators, manufacturers, and regulators in coalition to assure that type of incident never happens again.

Of course, what did happen was an aircraft was operated past its original design life objectives for flight hours and cycles, and the failure of the upper fuselage skin produced a spectacular example of what can happen if fatigue cracking isn't found and repaired when it initially occurs. The aircraft landed safely with one fatality, and that outcome speaks to the structural integrity of Boeing's original design for the 737 aircraft, as well as superb airmanship by an experienced flight crew.

That summer, the FAA hosted the first of these international aging aircraft conferences, and the result was what became the Airworthiness Assurance Task Force, or AATF, which was chartered to be the forum for operators, manufacturers, and regulators to address the system problems that culminated in the Aloha incident.

The AATF had eight original objectives in its charter, and it's worth reviewing what they were, because they are still driving the work of the AATF's successor organization.

**First**, continue to use our present system of maintenance and inspection with diligence and thoroughness. Find out why a single aircraft suffered major structural failure and adjust the system as necessary.

- This charge led to the formation of a structures working group for each type and model of aircraft which was close to exceeding its original design objectives for flight hours and landing cycles, and implement structural modifications in any area where missed or improperly conducted inspections could render the aircraft unairworthy.



- The original work of these structures groups is largely complete, and the task groups are now engaged in annual reviews and other special programs to make sure the airplanes continue to operate with the highest practical degree of airworthiness.

**Second**, initiate research to find better ways to assess structural condition and detect structural problems.

- This charge is the heart of the AATF's support for the wide-ranging research and development effort which is now underway at the FAA's Technical Center and at NASA.
- We owe a great deal to Congressmen Oberstar and Lewis for their support of the funding which is making this research possible. The results of these research efforts will begin to reach the field in the mid 1990's.

**Third**, pursue the concept of teardown and fatigue testing of the high time aircraft.

- Resulted in specific recommendations for audits.
- Now being written into rule form.

**Fourth**, pursue the transfer of currently available knowledge of NDT and its application to aircraft inspection.

**Fifth**, put R&D money into improving NDT techniques and methods.

- Basis for congressionally-mandated centers of technical excellence at a number of colleges and universities, as well as the Sandia National Laboratories.

**Sixth**, human factors

- Genesis of major human factors research effort for training, task qualification, task accomplishment.
- Major research efforts underway, including cooperative venture with USC and Continental Airlines.
- Continuing to track further efforts through AATF's and R&D working group.

**Seventh**, ensure that the communications systems between airlines, manufacturer, and FAA are adequate.

- Working group has identified near-term improvements for FAA's reliability data system and is writing a proposed rule change to implement.
- Further effort underway to exploit the state of the art in digital information systems.

**Eighth**, establish task forces from the airlines, manufacturing industry, FAA, and NASA to continue work begun in this work shop.

Let me try to describe for you what this effort has entailed.

The eleven structures working groups account for several hundred of the finest aeronautical engineers in the world. They have been meeting frequently since June 1988. The other working groups of the AATF involve the industry experts in each field. In addition, the FAA has established national resource specialists and technical oversight groups, as well as their advisory committee structure, to participate in this massive venture.

Airworthiness Directives for structure and corrosion have been developed for all eleven aircraft types and are now being mandated through rulemaking action. These AD's are major additions to the maintenance programs of the commercial airlines, and will cost millions of dollars per airplane to accomplish.

- Other task groups have addressed maintenance program guidelines, fatigue, and repair durability.
- The investment required to continue to operate older aircraft is unprecedented.

I would like to go back to the Aloha incident and re-examine for your benefit the things that failed and the system failures that allowed the Aloha incident to occur.

- There was inadequate on-site inspection of the aircraft; there was inadequate aircraft conditions surveillance;
- There was inadequate maintenance program oversight;



- There was inadequate training on the part of the mechanics and inspectors;
- There was inadequate respect for the need to provide adequate out of service and elapsed time to do maintenance work;
- There was an indication there were inadequate technical resources applied to the maintenance program;
- There was inadequate concern for the implications of the cold-bond failures; and there was an unwillingness for the manufacturer to raise the issue that this operator was not doing adequate inspections on this aircraft.

None of the above makes a statement about the age of the aircraft, which, incidentally, is why we changed the task force name to Airworthiness Assurance from Aging Aircraft.

The efforts of the AATF have resulted in some good actions which will marginally improve safety, but the most important results of these efforts were the cooperative effort that brought the true issues out as a result of having the regulatory authorities, the manufacturers and the airlines gaining a consensus on what was required to be done.

- It has demonstrated that cooperative consensus can result in faster action and better solutions.
- It demonstrated the need for significant changes in the rulemaking process.
- It demonstrated the need for speed. We are fast approaching an era of instant data and we must all adjust the infrastructure and organizations to capitalize on these changes.
- It also has demonstrated that we need to do more work on achieving consensus on priorities of safety issues.

When deregulation came along, only 26 percent of the American public had ever been on a commercial airlines. Today, that number is somewhere between 80 and 90 percent. The incredible growth that we have seen since

1978 was a result of flying becoming affordable. A recent study released has indicated that the growth in airline traffic resulted in a 4% reduction in city-to-city automobile traffic and that this 4% was saving 2,000 lives a year.

- Ladies and gentlemen, this is ten times the average life lost by all the airline incidents. Whether or not you accept those numbers as being accurate, you must accept that it is relative and that airline safety and airline traffic will certainly have an effect on other areas. When the history books are written, we may not fare too well because we have yet to determine how to assess relative priority with the efforts we are expending to improve safety, and we have not yet learned to quantify the relative worth of expending those dollars and energies on to an aircraft safety related item as compared to safety in other areas.

Since the Aloha incident there has been an incredible increase in focus toward the engineering and maintenance side of air transportation.

- AD's have proliferated almost 300%.
- MEL control has been tightening in several ways. First, we now have time limits specified - which was necessary. Second, MEL's have steadily become more restrictive. Third, airline authority for dispatching aircraft with conditions not significant to safety or airworthiness is being systematically removed.
- Since the creation of Part 121, airline engineers have had the authority to make minor changes to their manuals. Our DER's were removed because the FAA said with DAS/authority we didn't need them. Now, we're being told that any change to SRM's and engine maintenance manual must have explicit FAA or DER approval. Bottom line - any change to these manuals is now considered major.
- Major/minor has been a debate since Hector was a pup -

Collectively, we have addressed all nine of the deficiencies that culminated in the Aloha incident. From this point forward, we need to carefully study the cost benefits of any further conservatism; the public pays the bills and the public suffers the delays.



Collectively, we must:

- Devise a more rapid system for resolving issues like the MEL and major/minor.
- Allow airlines with engineering resources to make necessary changes to their manuals quickly.
- Provide more emphasis on the quality of decisions made and data used in justification rather than debating interpretation of vague FAR's.
- You managers, engineers and technicians devoting efforts toward aging aircraft issues must focus on the vital few issues.
- Continue to nurture the infant steps taken by the AATF effort toward a more cooperative working relationship between airline, manufacturers, and regulators.

Finally, we must do a better job of educating our folks, the congress, and the public on safety issues and the trade-offs involved in expending money and effort to improve safety.

We have one of the safest industries in the world and the U.S. system is clearly the best of the best. What industry can boast of a 60 fold improvement over a 30 year period. We've done a great job and I'm convinced, with cooperation, we'll continue the trend.

## AGEING AIRCRAFT - THE AATF PRESENT PROGRAMME

DON JAMES

(Head, Structures and Materials Dept, UK Civil Aviation Authority)

### JAA REPRESENTATIVE ON AATF STEERING COMMITTEE

#### Introduction - What is JAA?

As I am here representing JAA and many of you, being naturally more concerned with engineering and maintenance aspects of aircraft structures than the niceties of foreign regulatory systems, it is perhaps worthwhile to take a few moments to outline what JAA is.

In the early 1970's, a small number of European Airworthiness Authorities started work on the development of a common European Type Certification code for transport aircraft. The Regulatory Authorities participating in this work are known as the Joint Airworthiness Authorities (JAA).

Initially, the pressures for such work were purely economic and were mainly, if not exclusively, driven by aircraft constructors who were anxious to avoid the burden of designing aircraft to take account of differing standards in different countries; they were, and are, equally anxious to avoid certificating each aircraft design separately in each European country. Operators were, and are, equally anxious for common standards for, without them, transfer of aircraft from the register of one country to the register of another could result in the need for modifications to the aircraft to be carried out.

Since these early beginnings, the group of countries participating in this work has grown to form a group, known as the *Joint Aviation Authorities* (JAA), of 19 countries who have signed "Arrangements" documents and who manage their business of safety regulation (operations and airworthiness) through the JAA Committee. The original 1979 Arrangements represented a commitment jointly to develop common airworthiness regulations, known originally as Joint Airworthiness Requirements, so as to facilitate the development and certification of joint projects and to ease the import and export of aeronautical products. Since then, the joint regulations have extended from design to maintenance and now to operations; to reflect this wider range of interests the joint regulations are now called Joint Aviation Requirements (JARs).

Thus, JAA is developing into a comprehensive aviation regulatory system which will apply common requirements and procedures for all aspects of civil aviation (design and certification, maintenance and operation) throughout the 19 countries listed below:

#### Membership of JAA

Austria	Iceland	Portugal
Belgium	Ireland	Spain
Denmark	Italy	Sweden
Finland	Luxembourg	Switzerland
France	Netherlands	United Kingdom
Germany	Norway	Yugoslavia
Greece		

In order to carry out the functions of developing these requirements and procedures, JAA has a number of technical Study Groups; one of these is the Structures Study Group, of which I am chairman; this explains how I came to be nominated to represent JAA on the AATF Steering Committee.

## **Introduction - Work of AATF**

As it is now a well established part of our history, in the aftermath of the Aloha Airlines Boeing 737 accident on April 28, 1988, the FAA hosted the first International Conference on Ageing Aeroplanes.

One of the major outcomes of this first Ageing Aircraft Conference was the following set of joint Airline/Manufacturer recommendations:

1. Continue to use our present system of maintenance and inspection with diligence and thoroughness. Find out why a single aircraft suffered major structural failure and adjust the system as necessary.
2. Initiate research to find better ways to assess structural condition and detect structural problems.
3. Continue to pursue the concept of teardown of oldest aircraft to determine structural condition, and conduct fatigue tests of older airplanes.
4. Pursue transfer of the currently available body of knowledge of NDT and its application to aircraft inspection.
5. Put R&D money into improving NDT techniques and methods.
6. Examine all aspects of Human Factors involved, including training and qualification of airline inspectors.
7. Ensure that the communications systems between airlines, manufacturers, and the FAA are adequate.
8. Establish task forces from the airlines, manufacturing industry, FAA, and NASA to continue the work begun in this workshop.

ATA offered to initiate the process of forming a committee to star in the industry efforts towards implementing the above recommendations and to carry forward a programme aimed at identifying and eliminating potential issues of concern about the structural integrity of ageing aircraft.

The first meeting of the Airworthiness Assurance Task Force Steering Committee was held in Washington DC on August 12 1988. The composition of the steering committee was determined and Task Group efforts were identified.

Now, three years later, there have been some changes to the Steering Committee membership but the composition still retains the original emphasis on international membership with representation from Operators, Manufacturers, Regulatory Agencies and other interested bodies. The current Steering Committee Membership is:



Ade Hale	American Airlines - Chairman
Roger Casterline	United Airlines
Fred Schubul	Alaska Airlines
Julian May	Delta
Ron Wickens	Federal Express
"Sandy" Sell	IATA (British Airways)
Jack McGuire	Boeing
Amos Hoggard	McDonnell Douglas
Roger Lecomte	Airbus
Peter Bashford	AECMA (British Aerospace)
Sam Venneri	NASA
David Wadsworth	PAMA
Joe Schwind	ALPA
Leroy Keith	FAA
Bill White	FAA
David James	JAA (CAA - UK)
Joe Vreeman	ATA
Steve Erickson	ATA

As you can see, the Steering Committee is a unique body with membership representing a wide range of international interests.

The work of the AATF (which has recently been renamed AAWG to reflect its new status as a Working Group within the structure of the recently formed Aviation Rule Making Advisory Committee) has focussed on the following major issues:-

- Termination of structural service bulletins
- Corrosion prevention and control programmes
- Review of Supplemental Structural Inspection Programmes
- Maintenance programme design
- Repair assessment and documentation
- Research and Development
- Communications
- Fatigue Test and Tear Down Issues

For each activity, appropriate task groups were established with joint manufacturer/operator/regulator membership; in the case of Task groups dealing with aircraft model specific issues, only the specific aircraft constructor was involved; in all other cases a mix of constructors was involved. Overall, thirty two US and 52 foreign airlines, six aircraft constructors, and airworthiness authorities from 5 countries and numerous other organisations and research agencies have participated in the programme.

The achievements of AATF are too many to discuss in detail here and so I will concentrate on what I consider to be the major issues.

#### **Mandatory Modifications and Terminating Action**

Up to the time of the Aloha accident, the normal process for controlling the structural safety of "*fail safe*" or "*damage tolerant*" aircraft structures relied almost entirely on repetitive inspections. When a constructor/airworthiness authority became aware of a structure problem, such as fatigue cracking, which, if not detected and repaired, had the potential to cause a significant degradation in airworthiness, the normal practice was to introduce a service bulletin which defined inspection procedures (inspection method, threshold and repeat interval) which were designed to ensure with a high (but undefined) degree of probability that the structural damage would be detected (and be repaired) before a significant degradation in structural airworthiness occurred. Frequently these service bulletins also specified a modification/rework procedure that would eliminate the cause of the cracking problem and

provide an alternative to repetitive inspection as the means for ensuring continued structural integrity. In most, if not all, of these cases, the inspection part of the service bulletin has been made mandatory by means of an Airworthiness Directive.

The net result of the above procedure has been that the means of controlling structural airworthiness in the case of known fatigue cracking problems has been to carry out repetitive inspections of each and every affected aircraft until the fatigue damage is detected, and then to carry out a repair. Thus, the process has been totally dependent on repetitive inspection.

Following the Aloha accident, in which inspections failed to detect a known problem before catastrophic failure, and the discussion at the first International Ageing Aircraft Conference, it was decided that the policy of relying on repetitive inspections should be reviewed.

Consequently, AATF established working groups to review all known structural fatigue problems on:

Airbus	A300
Boeing	707
Boeing	727
Boeing	737
Boeing	747
British Aerospace	BAC 1-11
Fokker	F28
Lockheed	L1011
McDonnell Douglas	DC-8
McDonnell Douglas	DC-9
McDonnell Douglas	DC-10
McDonnell Douglas	MD-80

with a view to selecting these problem areas (i.e. Service Bulletins) for which action should be taken to avoid continued reliance on repetitive inspection as the primary means of airworthiness control.

Aircraft under specific working groups were established to post their experience and knowledge in order to select service bulletins for which the requirement for continued repetitive inspections should be terminated by making the incorporation of modifications which eliminate the source of each respective cracking problem mandatory.

The service bulletins which were selected were considered on the basis of:

Potential for a Safety Problem  
High Probability of Occurrence  
Difficulty of Inspection

As a result of this activity, Airworthiness Directives have been, or are about to be, issued to make the incorporation of these modifications, or other terminating action, mandatory. For the foreign manufactured aircraft in this programme, this works through the foreign airworthiness authority responsible for the type first issuing its own AD and then formally notifying FAA (and all other countries with the affected aircraft on their register) that they have done so. For many countries (including the UK) an AD applicable to a foreign manufactured aircraft issued by the authority of the state of manufacture is normally automatically applicable and so no further AD action is necessary. In the case of the United States, of course, the legal procedure requires that the subject matter of the foreign AD is subject to all the normal AD rule making procedures.

The current status of these Structural Modification ADs in the United States is as follows:

## STRUCTURAL MODIFICATION AIRWORTHINESS DIRECTIVES

<u>PRODUCT</u>	<u>AD NUMBER</u>	<u>EFFECTIVE DATE</u>	<u>COMMENTS</u>
Boeing Model 707/720	91-07-19	April 29,1991	
Boeing Model 727	90-06-09	April 17,1990	
Boeing Model 737	90-06-02	April 17,1990	
Boeing Model 747	90-06-05	April 17,1990	
Douglas Model DC-8	90-16-05	September 10, 1990	
Douglas DC-9/MD-80	90-18-03	September 24, 1990	
Douglas Model DC-10	90-16-04	September 10, 1990	
Lockheed L-1011	91-05-05	March 22,1991	
BAe BAC 1-11	90-23-09	December 10,1990	CAA AD: 003-12-89, 21 December 1989
Fokker F28	91-05-10	March 25,1991	RLD AD: BLA 90-063 effective July 1, 1990
Airbus A300	NPRM	Comment period closed	FAA AD expected mid November 1991 DGAC AD: CN 90.222.116B, issued 22 December, 1990

Work in this area has not stopped with the publication of these Airworthiness Directives; annual reviews are being carried out in order to ensure that any new problems that need terminating action are dealt with in a similar way.

### Corrosion Prevention and Control

Ageing aircraft surveys by Major manufacturers, operator comments at the first Ageing Aircraft Conference, and defect reports to airworthiness authorities all indicated that corrosion continues to be a significant problem, world-wide.

This is a major concern as inadequate control of corrosion can result in a major degradation of structural integrity and can result in the catastrophic loss of an aircraft.

In some respects, corrosion is not purely an ageing aircraft issue because, in many cases, if proper precautions are not taken, corrosion can, and frequently does, cause significant structural degradation in a remarkably short time.

However, with ageing aircraft, corrosion becomes an even more significant issue because:

As aircraft age, widespread corrosion is more likely to exist.

As aircraft age, fatigue cracking is more likely to exist and so combination of corrosion and fatigue cracking becomes more likely.



Not enough is known about the interaction of fatigue and corrosion.

For these reasons, it was decided by AATF that baseline Corrosion Prevention and Control Programmes should be established for each aircraft model in order to define the minimum requirements for assuring fleet wide prevention and/or control of all corrosion problems that might affect the continuing airworthiness of the world-wide fleet.

In order to do this, an Industry wide committee was set up to define and co-ordinate policy and the model specific task groups were given the task of defining the minimum requirements for each aircraft model.

The outcome of this activity is a set of new Corrosion Prevention and Control Programmes which take into account the world-wide experience of each aircraft model but which are written, as far as is practicable, in a common format using common definitions, for example, of corrosion severity.

The final action is to make compliance with these new baseline Corrosion Prevention and Control Programmes, or equivalent, mandatory. The FAA's initial intention was to do this by Airworthiness Directive, as for the modification programmes; the current state of this activity is:

#### CORROSION PREVENTION AND CONTROL AIRWORTHINESS DIRECTIVES

<u>PRODUCT</u>	<u>AD NUMBER</u>	<u>EFFECTIVE DATE</u>	<u>COMMENTS</u>
Boeing Model 707/720	90-25-07	December 31, 1990	
Boeing Model 727	90-25-03	December 31, 1990	
Boeing Model 737	90-25-01	December 31, 1990	
Boeing Model 747	90-25-05	December 31, 1990	
Douglas Model DC-8	NPRM	Comment Closed	FAA AD in final coordination
Douglas DC-9/MD-80	NPRM	Comment Closed	FAA AD in final coordination
Douglas Model DC-10	NPRM	Comment Closed	FAA AD in final coordination
Lockheed L-1011	NPRM	Comment Closed	FAA AD in final coordination
BAe BAC 1-11	NPRM	Comment Closed	FAA AD in final coordination CAA AD 021-11-90 issued November 1990.
Fokker F28	NPRM	Comment Closed	RLD AD: BLA 90-051 effective June 14, 1991
Airbus A300	NPRM	Comment Closed	FAA AD in final coordination DGAC AD: CN 90.196.117B, issued 14 November, 1990

Since the publication of the Corrosion Prevention and Control AD's on the Boeing aircraft, a number of issues have arisen in the United States about the use of the AD for such a purpose and about the reporting requirements contained in the AD's, and, also relative roles of the PMI and Aircraft Certification Office. It is expected that these

issues will be resolved by FAA in the very near future and that publication of the AD's on the remaining aircraft listed will follow shortly after.

### **Fatigue Test and Tear Down Issues**

During the investigation of the Aloha accident, there was a great deal of discussion about the potential dangers of the multiple site fatigue damage problem.

Many commentators expressed the view that reliable detection of multiple site fatigue damage in airline service requires inspection methods which are beyond the current state of the art. Consequently, such commentators believe that the only safe way to contain multiple site fatigue damage is to avoid its occurrence in service.

This point of view is illustrated by recommendation A-89-67 of the NTSB report on the Aloha Airlines accident. One of their recommendation to FAA is as follows:

(FAA should) require that all turbojet transport category airplanes certificated in the future, receive full scale structural fatigue testing to a minimum of two times the projected economic service life. Also require that all currently certificated turbojet transport category airplanes that have not been fatigue tested to two lifetimes, be subjected to such testing. As a result of this testing and subsequent inspection and analysis, require manufacturers to identify structure susceptible to multiple site damage and adopt inspection programs appropriate for the detection of such damage.

In response to this recommendation, and similar pressures from other influential sources, together with its own analysis of the situation, FAA started a rule making project for an SFAR which *"would require full-scale fatigue testing on the current fleet of ageing transport airplanes to two times their design service life. If adopted this rule making would apply to all turbine powered transport category airplanes certificated in any category"*.

Because of the significant impact that such a rule, if adopted, would have on the operation of many, if not most, of the transport aircraft in operation today, AATF established an industry wide subcommittee to review the relevant issues.

The Subcommittee was commissioned by the Airworthiness Assurance Task Force in June 1990. It was formed to consider the merits of Airline/Manufacturer Recommendation Number 3 from the International Conference on Ageing Airplanes (see introduction). This recommendation stated:

- "3. *Continue to pursue the concept of teardown of the oldest airline airplane to determine structural condition, and conduct fatigue tests of older airplanes per attached proposal."*

The AATF-SC further asked that special emphasis be placed on the ability of the concept to determine the onset of multiple site damage or other widespread structural degradation.

The subcommittee's objective was to determine the benefits/limitations of teardown or fatigue tests to enhance the confidence of continuing airworthiness. Based on this assessment, and considering other available options and actions in work, establish an industry common position.

Being a duly authorised Subcommittee of the AATF, participation from all sectors of the industry was invited. Representatives of the following organisations participated:

### Fatigue Test and Teardown Sub-Committee Membership

Airline Operators	Manufacturers	Regulators
American	Airbus	CAA (UK)
Delta	British Aerospace	FAA (USA)
Lufthansa	Boeing	LBA (Germany)
Pan Am	Douglas	RLD (Netherlands)
United	Fokker	
	Lockheed	

In the course of its work, the subcommittee reviewed the ageing aircraft concerns which had been perceived to exist in 1988 and the work that was being done, mainly under the guidance of AATF, to address these concerns.

The subcommittee found that significant progress has been made to alleviate the concerns identified in 1988, through the development of the AATF programs for structural modification and maintenance improvements. However, some issues partially remain as concerns and further evaluation was considered appropriate to determine if additional actions are required. These remaining concerns are:

1. Industry confidence in the ability to predict and detect potential widespread fatigue damage in the future.
2. The need for further improvements in reporting, recognition, and action in service events (manufacturers, operators, and regulators).
3. The need for further training of operators and regulators to increase the awareness and knowledge of ageing airplane issues.
4. The influence of major structural modifications to original designs that are not the responsibility of the airplane manufacturers.

These concerns provided the background for consideration of various options to enhance the confidence in continuing structural airworthiness of older airplanes.

The Subcommittee reached seven major conclusions:

1. Major improvements in the structural safety system are being introduced by the AATF programs, including model specific reassessment of the Supplemental Structural Inspection Programs. These AATF programs contain elements to address all concerns existing in June 1988.
2. There is still an outstanding concern for the potential onset and possible non-detection of multiple site and multiple dependent element damage in the fleet. Until the AATF programs are fully developed, it will not be known if this concern is fully addressed for each model.
3. An AATF audit of these programs is appropriate for each model to determine if any specific areas of concern remain. Appropriate action should be developed, either through expansion/revision of the existing AATF programs or by additional measures.



4. In the event that additional data or fleet experience is required to validate the effectiveness of the AATF programs for each model, several viable options exist as a source of data either singly or in combination. They include:
  - A. Selected limited non-destructive disassembly, inspection and refurbishment of high time airplanes continuing in service.
  - B. Continuing assessment of the fleet demonstrated capability through diligent monitoring of service experience.
  - C. Fleet exploration of high time airplanes with improved state of the art Non Destructive Inspection (NDI) techniques:
  - D. Testing of new or used structure on a smaller scale than full component tests (i.e. sub-component and/or panel tests).
  - E. Fatigue test of high time airplane or full scale major component followed by detailed teardown of test article.
  - F. Teardown of high time airplane.
5. Tests, teardowns, and analysis can provide valuable data and insight into probable future behaviour, however they can never replace a conscientious and reliable structural inspection program. The manufacturers, operators and regulatory agencies must continue to play their vital roles in developing and monitoring these inspections.
6. The use of advanced, reliable and user friendly NDI techniques to supplement visual inspections methods is of paramount importance in the effective inspection of large fleets of airplanes.
7. The Subcommittee has concentrated on only one of the four perceived concerns existing in 1990, that of potential widespread fatigue damage. This Subcommittee believes, however, the remaining concerns discussed on Section 5.0 are equally important.

Based on the conclusions of this sub-committee, the following recommendations were made:

1. That the AATF direct the working groups to perform an audit (as described in the Sub-Committee's Report) of all airplane models that have exceeded or are within five years of exceeding their respective original design life goal. The audit should be accomplished with respect to the AATF programs to determine if all concerns are adequately addressed and if any additional action is appropriate.
2. That the manufacturers implement a long term cooperative program to develop a common understanding of multiple site damage and multiple element damage phenomena. They should work towards a common approach to identifying likely locations, consequences and assessment techniques to ensure maximum protection from such damage in existing and future designs.
3. To ensure consistency, a technical oversight group should be established to review the findings of Recommendations 1 and 2. This oversight group, which should include appropriate manufacturer/regulator technical representation, would report to and provide periodic reports to the AATF-SC.
4. That the AATF initiate discussions with the FAA and TOGAA to gain their support for the above recommendations as an alternative to any potential unilateral rule making activity. These

discussions should be in the cooperative spirit typical of AATF activities and should occur before any initiatives or mandated processes are implemented.

5. That increased emphasis be placed on the near term development of advanced, reliable NDI techniques by NASA, FAA, and the industry.
6. That the AATF ensure that adequate programs exist to address the remaining concerns (outlined above).

A new task group, known as the *Structural Audit Evaluation Task Group (SAETG)*, which has virtually the same membership as the sub-committee which produced the above recommendations, is now actively involved in defining procedures for implementing the Audit defined in the above recommendations.

### **Repairs:**

Early in the AATF programme, it was recognised that the long term damage tolerance of structural repairs is a potential safety concern for ageing aircraft.

Although most structural repairs have performed properly in service, it has been recognised that much of the effort going into the ageing aircraft programme has the effect of a re-evaluating the structural damage tolerance to the standards of FAR 25 amendment 45, no such effort has been going into the re-evaluation of structural repairs, apart from specific action by manufacturers in particular cases.

Many, if not most, of the structural repairs that have been carried out over the years (except, perhaps, for the recent past) have been designed with the sole objective of ensuring that the static strength of the repaired aircraft is as good as, and better than, the original strength.

However, repairs designed with such an objective may have the effect of degrading damage tolerance as a result of degrading inspectability, or fatigue life, or both.

In order to review and address these concerns, AATF established yet another industry wide subcommittee which identified the need for an operator usable system which could be used to categorise the damage tolerance of existing structural repairs.

Following the precedent set by the Corrosion Prevention and Control Sub-committee, the Repairs Sub-committee set out to define programmes which, as far as practicable, would be similar across the industry in format and data requirements and which would allow the operators to do the necessary categorisation.

The programme, which will involve operators in a substantial amount of work through the need to:

- visually inspect each aeroplane for repairs

- chart locations of repairs on plating, wing etc degradation

- conduct records search

- use manufacturer provided documentation and methodology to classify repairs

has already involved the Constructors in a substantial amount of technical work in developing methods which will enable them to produce procedures capable of implementation by operators which will enable the damage tolerance classification of repairs to be carried out. The detailed work of the supporting manuals is substantial.

However, the repair issue has a number of facets which involve a number of regulatory issues, some of which extend beyond the remit of AATF and includes such issues as:

SFAR 36 Authority  
Major/minor classification  
OEM responsibility for repairs  
Third Party repairs  
Damage Tolerance  
Effect of repairs on SSID

In view of the wide ranging nature of some of these issues, it has now been decided a survey of fleet repairs should be carried out before deciding on the future course of action on the repair effort. Initially, a preliminary sample survey of large fuselage repairs located below the window line will be carried as a precursor to developing a plan for a repairs assessment survey for all operators world-wide.

The purpose of the world-wide survey will be to determine whether the world-wide fleet condition for such repairs warrants further action.

An example of the type of unacceptable repair that has recently been found in the UK is shown by figure 1. This particular repair, which was above the window line, was found by a UK Maintenance Organisation when working on a non-UK aircraft.

#### **Conclusions:**

Some of the achievements which have been outlined here represent major significant changes of policy, for example:

Mandating terminating action instead of continued reliance on repetitive inspection.

The development of baseline corrosion prevention and control programmes which are based, through the involvement of the aircraft constructor and the certification authority, on world-fleet experience.

Other issues, perhaps equally important in the long term, have not been mentioned through lack of time:

Review of Supplemental Structural Inspection Programmes

New Structural Maintenance Programme Guide-lines for Continuing Airworthiness

Proposals for changes to the Service Difficulty Reporting Regulations

Review of the FAA Ageing Aircraft Research Programme by the Research and Development Subcommittee.

Since it was formed in August 1988, the AATF, now AAWG, has demonstrated a remarkable achievement of international collaboration between Operators, Constructors and Regulators in working together to solve problems of mutual concern. Significant progress has been made on most issues, although the final resolution of some has yet to be reached.

We can all be sure that AATF and those who have participated in its work have made, and are continuing to make, a major contribution to the continued airworthiness of ageing aircraft.



# Sketch of FUSELAGE LAP JOINT

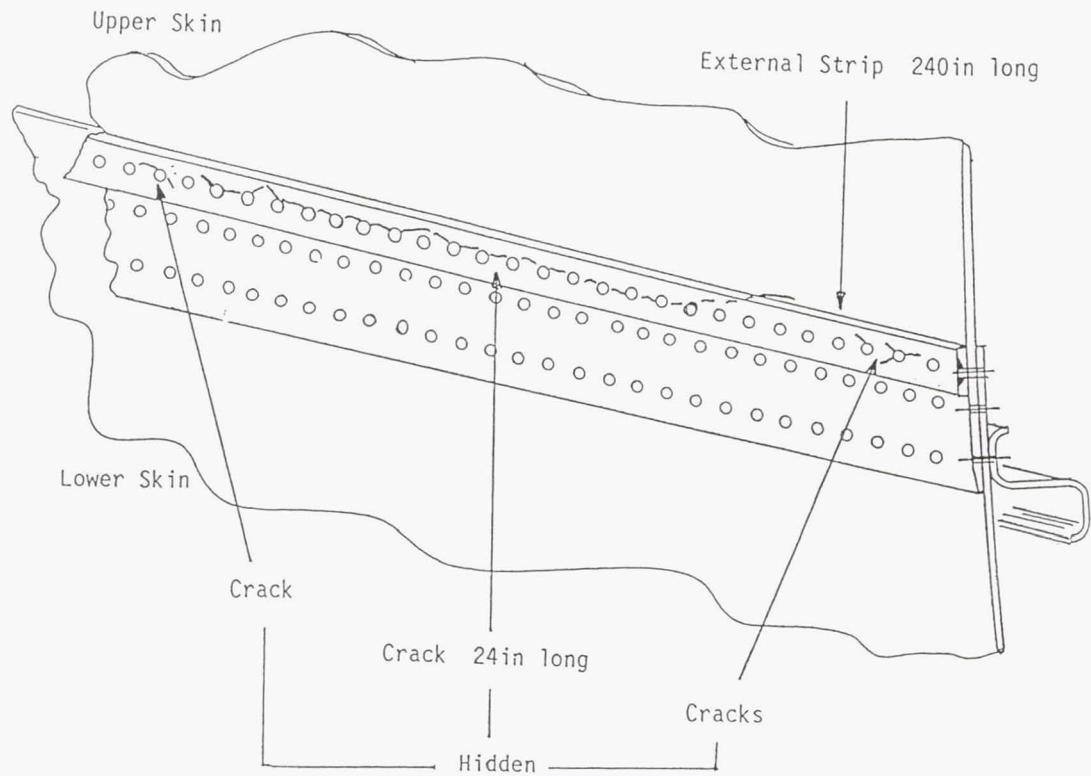


Figure 1

FEDERAL AVIATION ADMINISTRATION  
NATIONAL AGING AIRCRAFT RESEARCH PROGRAM

By S. G. Sampath  
FAA Technical Center  
Atlantic City, NJ

I am going to provide a progress report on the National Aging Aircraft Research Program that we at the FAA Technical Center are conducting, in concert with NASA and the Industry--the Program that was conceived in the form of a specific recommendation provided to the FAA during the First Aging Aircraft Conference that was held in Washington in June, 1988. The FAA immediately acted on the recommendation to establish a research program to find better ways to assess and deal with the structural condition of aging aircraft and commissioned us to conduct a study of various associated technical issues.

We then proceeded to develop a plan of research which was published in 1989. It is worth mentioning that since that time the FAA has been formally empowered by the Aviation Safety Research Act that was promulgated by the U.S. Congress in 1988, which states that "the (FAA) Administrator shall undertake or supervise research to develop technologies and to conduct data analysis for predicting the effects of aircraft design, maintenance, testing, wear, and fatigue life of the aircraft and on air safety, to develop methods of analyzing and improving aircraft maintenance technology and practices (including nondestructive evaluation of aircraft structures)". The language in a subsequently enacted Public Law requiring the Administrator to undertake or supervise research "...to develop technologies and methods to assess the risk of and prevent defects, failures, and malfunctions...which could result in a catastrophic failure of an aircraft" reinforces the legitimacy of the goals of the National Aging Aircraft Research Program.

Before recounting what has been accomplished since the previous aging aircraft conference held in Bordeaux in 1990 and report on the course of the program--a state of the Program, so to speak--my colleagues and I gratefully acknowledge the support and encouragement that we have been receiving from you in the aviation community, and are thankful for your advocacy for continuation of our efforts.

I begin by reporting to you that a revised National Aging Aircraft Research Program Plan has recently been published. The contents of the revised plan were identified during discussions with the offices responsible for certification of transport and small airplanes, and the offices which deal with aircraft maintenance and certification of engines and propellers --all within the FAA. The FAA National Resource Specialists were also consulted during the process. Likewise, meetings were held with the Airworthiness Assurance Task Force and its constituent Research and Development Working Group, with the Technical Oversight Group on Aging Aircraft and a recently constituted group of independent experts on nondestructive inspection. Their suggestions were carefully evaluated for incorporation into the Plan. Other recommendations made previously by the General Aviation Manufacturers Association and the Regional Airline Association were



also taken into account. During the past year, in the process of setting up cooperative research projects, the program objectives were discussed with some international regulatory authorities to gain their perspectives as well. Still, we feel that the process should continue to allow us to periodically tune the Plan to maintain its currency. Thus, we will be preparing and issuing revisions to the Plan, as needed.

I am pleased about our interaction with NASA in regard to aging aircraft research. During his presentation, Dr. Venneri will indicate the considerable progress that has been made towards unifying the FAA activities and those underway at NASA. In the main, the projects within the FAA program will address continued airworthiness issues while NASA, in keeping with their established charter and expertise, will focus on basic technology development. The cooperative agreement between FAA and NASA is bound to have a synergistic outcome.

During the Conference sessions yet to come, you will be apprised about several projects being conducted under the auspices of the Program. However, the broad issues shaping the research agenda are the following:

**Inspection:** Instances of accidental discoveries of long cracks in flight safety critical assemblies of airframes, even though some may argue that such cases are to be classified as isolated, point to the need for taking purposeful action to strengthen the aircraft inspection systems. The inevitability of aging structures being increasingly susceptible to fatigue cracking and corrosion related damage points to the same imperative. Where possible, assurance of airworthiness through inspection might be the preferred option provided visual acuity can be enhanced through auxiliary devices and Non-Destructive Inspection (NDI) equipment and procedures with higher reliability are available. Accordingly, the Plan emphasizes the need for improved NDI systems by stressing research and validation efforts aimed at fielding several prospective and alternative technologies that can be matched to specific inspection tasks.

**Fatigue:** Wide spread fatigue damage, also referred to as multi-site damage or MSD, is being encountered in several makes and model of airplanes. Because the ability of a structure to contain a fracture may be circumvented in the presence of wide spread damage, identification of design corridors that postpone MSD and development of strategies to treat structures affected by MSD are being prioritized.

**Commuter Airplanes:** Although the commuter airplane fleet does not seem to have been touched by untoward incidents, it counts among its ranks a number of airplanes that were built about 40 years ago. The engineering principles that were adopted for designing such airplanes have generally been superseded by methodologies that yield increased structural reliability. Similarly, more effective products to control corrosion are now available. Hence, development of analysis tools and data associated with the newer methodologies to evaluate fatigue quality are planned to facilitate their adoption by the Commuter Aviation sector, as are educational aids to promote awareness about effective corrosion controls and affordable methods for inspection.

**Support FAA Field Offices:** The FAA is preparing several important rulemaking notices pertaining to continued airworthiness to enhance structural safety. The research program is prepared to provide data to the appropriate certification offices to support their actions.



**Human Factors:** Understanding the factors in the current aircraft maintenance system which degrade human performance probably has the biggest pay-off potential for dealing with the aging aircraft issue because it will lead to identification of remedial measures that can be implemented in the short-term. We are working with FAA specialists in Human Factors and Aviation Medicine towards that end.

**Training:** Still in reference to structural airworthiness, the desirability of rejuvenating the technical infrastructure within the aviation community is presumed. In that regard, an ongoing project that seeks to develop specialized course curricula at various universities and the project which has been offering a series of specially designed corrosion training courses for maintenance inspectors exemplify the responsiveness of the Program to meet the goal.

**Innovation and Long Term Research:** Research to address anticipated future concerns is consistent with the pro-active stance of the FAA. A part of the program resources is being directed at projects to explore innovative ideas. Towards the same end, emphasis is being placed on development of models and procedures that will serve the design needs of future generation aircraft as well.

The six disciplines within the Program are airplane flight loads, structural integrity, mechanics of materials and corrosion, inspection methods, maintenance, and human factors. However, many projects are multi-disciplinary in content. At this point, a few of the projects we have on hand bear mentioning.

In reference to **flight loads** the data base describing the operational load spectrum which the airframe has to sustain is incomplete, especially on account of changes in air traffic control patterns. More importantly, knowledge about flight loads experienced by regional commuter airplane operators is practically non-existent. Consequently, the Program has launched a multi-year flight loads data acquisition project involving a number of large and small airplane types and flight conditions replicating typical fleet service.

**Structural integrity** related research includes a number of projects, both analytical and experimental, that relate to recently issued airworthiness directives and contemplated rule changes. For instance, the longer term effects of the terminating action mandated by the FAA for fuselage lap-splices in certain old Boeing 727s and 737s are being studied through an elaborate test program involving test articles that are generically similar. The results from the test program might well indicate that the strategy used for dealing with the 727 and 737 cases can be used for other situations of widespread fatigue damage which may arise. In another structural integrity related effort, we are inviting the airframe manufacturers to participate by analyzing their own experiences to establish the connection between service cracking and results from full-scale fatigue tests, component tests, small scale tests and analysis. Another example of industry involvement in our research pertains to user friendly software that can be used for designing repairs with superior fatigue quality. It is envisioned that repair stations and independent designated engineering representatives (DER) will be significant end users of the software.

In the **materials and corrosion** area, we will fill a long felt need for a materials property data base that can be shared by researchers throughout the world, by first organizing and consolidating currently available data. Also on the agenda is a study to rank the performance of currently



available corrosion inhibitors as a function of various corrosion modes and corrosion prone sites. A project that was initiated about a year ago has yielded interesting and useful results about fatigue crack growth in materials treated with commonly used products for corrosion control.

A comprehensive assessment of conventionally used **NDI methods** to determine their optimality with respect to specific inspection tasks has been completed and the suitability of emerging NDI candidate systems in reference to field conditions is ongoing. A recent FAA-sponsored test that was performed at a major airline facility, which demonstrated that reliable inspection and substantial cost savings can be realized through the use of one such modern procedure--namely, shearography--bears witness to the potential benefits that have started to accrue as a result of our Program. Several studies are investigating a range of advanced techniques that might be available in the longer term for detection of cracks, corrosion and disbonds, especially in the invisible, secondary layer of the structure. In recognition of the fact that robotics and automation can significantly enhance reliability and can be used to inspect areas that are otherwise difficult to access, development of related technology to be conjunctively used with NDI equipment is an active element of the program. A hangar facility to house a number of specimens and assemblies with flaws has recently been commissioned. That "proving ground" will be invaluable in demonstrating and validating new and contending NDI equipment.

A complex maintenance related project that is under way seeks to quantify the probability of crack detection as a function of the crack size in a real depot setting. Results from such tests are essential for performing probability based modelling which in itself is crucial for a rational derivation of system reliability indices and the prediction of risk of structural failure due to structural damage. Due to the latter reason, the project planning effort has attracted participation by a number of organizations including the Air Force. Also related to that project is a task being carried out by AEA-Harwell in the UK, under an international cooperative research agreement, to quantitatively measure human performance during inspection as a function of fatigue, boredom, and environmental factors.

Other projects that have been or are due to be initiated relate to fatigue life extension through processes designed to induce favorable stress distribution at critical locations. Developmental work is needed before such technology can be transferred to the Industry. In a similar vein, advanced fractographic techniques are being used and improved to allow accurate metallurgical analyses of fracture events.

I would like to mention one more project, termed as Intelligent Service(s) Network, that is being considered, which could make an outstanding contribution to improving communications between the regulators, manufacturers, airline operators, repair stations, etc. In its entirety, the project is a multi-year effort to create a system that would allow data residing in widely disparate forms and computer systems to be readily accessed for information vital to proper maintenance and safety related actions. An integral part of the project will examine what information should trigger concern about the health of the system. It is expected that a demonstration program which will contain the required link, data transfer and secured access capabilities will be available within two years.

The FAA and NASA consider outreach activities to publicize the results from their Programs to be crucial to their success. Organization of conferences such as this one, specialized workshops

and symposia, which on all the previous occasions have been well attended and attracted delegates from within this country and the international community, will continue.

Obviously, management of the complex and diverse elements within the Program is a challenging task. To meet the challenge, we have embraced the concept of technical teams which are led by FAA and NASA staff and composed of the principal investigators and other technical experts to meet regularly to discuss merits of one another's approaches and to review results. Two such meetings pertaining to the subjects of multi-site damage and probability based failure models have taken place and a meeting to discuss corrosion fatigue related projects is scheduled to be held in Pittsburgh. Meetings pertaining to activities involving other technical subjects within the Program are in the planning stage.

The technical resource base for implementation of the Program has already been significantly broadened. We now have academic institutions, contract research organizations, aircraft manufacturers, airline operators, repair stations, material suppliers, the centers at Iowa State University, Sandia Laboratories and Wichita State University, other government agencies within the Departments of Transportation, Defense and Energy, and international establishments working on various aspects of the Program. The participation of the international establishments is made through formal agreements with the FAA counterparts. Some cases, like the cooperative research agreements with the Civil Aviation Authority in the United Kingdom and the Dutch regulatory body involve joint funding. In some others, currently being negotiated, the host country will assume the entire responsibility for funding.

The FAA technical capabilities are also being expanded by selective hiring of highly qualified people. We are also planning to invite world class scientists from within the country and abroad to work on a temporary basis alongside our staff to participate in the technical facets of the Program. Implementation of a plan to develop appropriate in-house test facilities is due to begin shortly.

In conclusion, the National Aging Aircraft Research Program is dynamic and vibrant, and you in the aviation community can expect potentially useful results and tools to continuously flow towards you, enabling a safer aviation fleet. Your past participation has been the key to the formulation of the Program plan and your continued participation in channelling our research towards solution of relevant technical issues is essential for our reaching your expectations.



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# Structural Integrity of Future Aging Airplanes

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A multitude of design considerations is involved in ensuring the structural integrity of Boeing jet transports that have common design concepts validated by extensive analyses, tests, and three decades of service. As airplanes approach their design service objectives, the incidences of fatigue and corrosion may become widespread. Continuing airworthiness of the aging jet fleet requires diligent performance from the manufacturer, the airlines, and airworthiness authorities.

Aging fleet support includes timely development of supplemental structural inspection documents applicable to selected older airplanes, teardown inspections of high-time airframes retired from service, fatigue testing of older airframes, and structural surveys of more than 130 airplanes operated throughout the world. Lessons learned from these activities are incorporated in service bulletin recommendations, production line modifications, and design manual updates. This paper gives an overview of traditional Boeing fleet support activities and the anticipated benefits for future generations of commercial airplanes based on the continuous design improvement process.

## BACKGROUND

This overview presentation is intended to focus more emphasis on the next generation of aging jet transports. These airplanes will benefit from lessons learned from current aging fleet programs both in terms of design improvements and maintenance-related issues.

Diligent attention to detail design, manufacturing, maintenance, and inspection procedures of the last three decades have produced long-life damage-tolerant structures with a credible safety record. The primary cause



of hull loss accidents is attributed to human factors and weather with about 10% attributed directly to the airplane, systems, or propulsion components or how they were maintained. Approximately 3% of the hull loss accidents are caused by airplane structure failures.

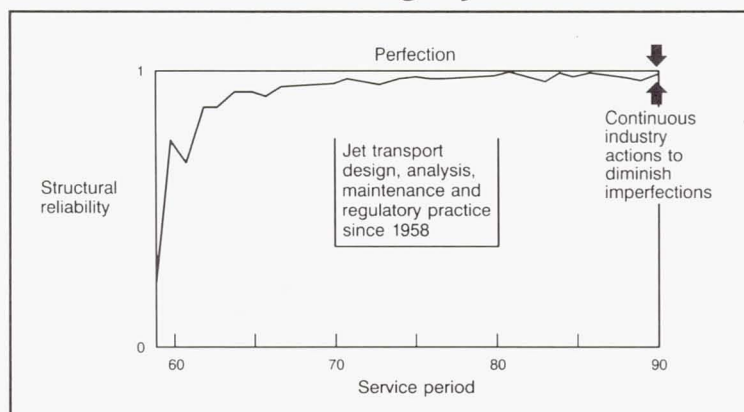
## EXPERIENCE RETENTION

In 1971, The Boeing Company initiated the Structures Durability Program. The program was a major step in experience retention and the formulation of an engineering approach to fatigue detail design. A summary of discrete types of fatigue distress and structural areas prone to repetitive problems was generated.

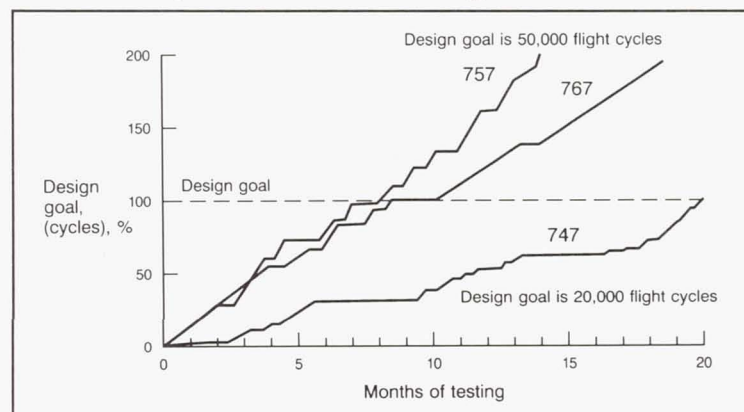
The knowledge gained provided a comprehensive understanding of detail design quality and became the foundation for building an improved fatigue design methodology. Fleet comparisons, additional testing and teardowns of high-time service airplanes added confidence in these new methods. Five years after the inception of this program, Boeing's Design for Durability manuals was made ready for application to new airplane design.

The durability methods used during development testing and interpretation of test data provided the confidence to proceed into the production design phases of the 757/767 airplanes at operating stress levels higher than those used on previous models. The effectiveness of the total design effort, however, was demonstrated by full-scale airframe fatigue tests. Both the 757 and the 767 airframe structures were subjected to comprehensive flight-by-flight fatigue tests. Major airframe inspections were conducted at intervals approximately equivalent to 10,000 flights. Visual inspections of the entire airframe, like those used by airlines, were performed. Inspection locations and procedures followed the maintenance planning data supplied for certification. After completion of 50,000 flights, a major inspection was conducted with loading fixtures and major reaction straps removed. After completion of 100,000 flights, a final teardown inspection was conducted. The fatigue test results of both the 757 and 767 major airplane tests were exceptional as proven by the rapid completion of 40 years (100,000 flights) of testing in record calendar

## Structural Integrity Record



## Major Airframe Fatigue Tests





time, without any down-time caused by structural fatigue cracking. The Structures Durability Program goal of reducing up to 85% of detail design problem areas was far exceeded.

## RECENT TESTING

### Test Verification

It is impractical to conduct verification testing for all critical loading conditions and portions of the airplane structure. Analysis methods and allowables verified by test are therefore used for airworthiness certification. Verification tests comprise small laboratory test specimens; large panels; major components representing wing, empennage, and fuselage structure; and full-scale airframes.

Full-scale static testing of new models is conducted to verify limit load carrying capability and satisfy certification requirements. In addition, full-scale fatigue tests are conducted to locate areas that may exhibit early fatigue problems. This allows necessary modifications of details that might exhibit early cracking and to demonstrate compliance with fatigue design objectives. Full-scale fatigue tests are also useful to develop and verify inspection and maintenance procedures. Fatigue tests are not intended to demonstrate "safe life" limits of structures certified as damage tolerant, nor are they an alternative to the inspections required for continued safe operation of aging airplanes.

Testing of new airplane structures does not incorporate corrosion and/or accidental damage that can accelerate fatigue cracking. Similar tests are conducted on older airframes to gain insight into the problems that might be experienced on high-time airplanes with repairs and service-caused defects.

### 747 Full-Scale Fuselage Testing

One of the Boeing initiatives to ensure continued safety and prudent maintenance of aging 747s was the testing of the fuselage of a retired 747-100 airplane that experienced 20,000 full pressure cycles. Extended cyclic pressure testing was conducted to evaluate the durability of the fuselage structure for operation beyond 20,000 full pressure cycles. A practical fuselage inspection

### Full-Scale Fatigue Tests

Model	Fatigue test cycles	Remarks
707	50,000	Hydro-fatigue test
727	60,000	Full airframe
737	130,000	Aft fuselage
747	20,000	Full airframe
	<b>40,000</b>	<b>Fuselage</b>
	<b>60,000</b>	<b>Nose section</b>
757	100,000	Full airframe
767	100,000	Full airframe

### 747 Fuselage Fatigue Tests

Used airplane (747-100SR) fuselage:



- Line position 229 delivered to Japan Airlines 12-21-73
- 14 years of commercial operation (20,000 equivalent full-pressure cycles)

Production (747-400) fuselage:



- Line position 697
- Fuselage Sections 41 and 42 (redesigned Section 41)

program was developed based on the cyclic test results, extensive damage growth data, and the results of the detailed teardown inspections of the skin splices. The inspection program is for the 747-100, -200, and -300 models and is in addition to the other aging fleet actions defined by the joint Boeing, airline, and FAA working group. Appropriately modified programs apply to specific long- and short-range versions. A 747-400 forward fuselage was also cycled to verify the effectiveness of the continuous design improvements based on fleet experience. The 747-400s will require much less maintenance for fatigue than the earlier 747s for operation past 20,000 full pressure cycles.

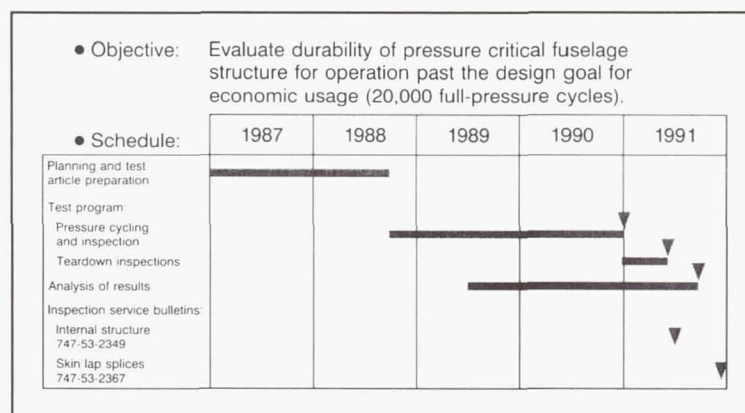
The planning for this program was initiated in August 1986. The cyclic pressure testing and teardown inspections of both test articles were completed during April 1991. The 747-100 airplane was delivered to the customer in December 1973 and was in commercial operation for 14 years. The airplane was received back at Boeing in April 1988. The wings and empennage structure were removed from this airplane and extensively disassembled for detailed inspections. After removal of the interiors, systems, and electrical components, the fuselage was moved onto the test slab. During the cyclic test phase, 20,000 full pressure cycles were applied for a total of 40,000 full pressure cycles on this fuselage.

The 747-400 forward fuselage test article came off the production line in April 1988. The cycle testing of this article consisted of two phases. During the durability evaluation phase, this test article was cycled to 40,000 full pressure cycles. Following that, for the damage growth test phase the test article was cycled an additional 20,000 cycles for a total of 60,000 full pressure cycles. Both test articles were subjected to detailed teardown inspections at the conclusion of the cyclic testing.

The three major objectives of this test program were:

- To evaluate the durability of the fuselage structure of the 747-100, -200, and -300 models for operation past the economic design objective (20,000 full pressure cycles).

## 747 Fuselage Fatigue Tests Test Program





- To evaluate the durability of skin lap splices, obtain crack growth data for multiple site damage scenarios on skin lap splices, and obtain damage growth data for internal fuselage structural components.
- To evaluate the durability of design improvements incorporated over the years and the -400 model-unique structural configuration items.

All the above objectives were achieved from the data generated by the two test articles.

The 747-100 fuselage test article was subjected to a thorough inspection prior to test cycling. Detailed visual inspections were performed on both external skin and internal structural components. The upper row of fasteners on all skin lap splices was inspected using high-frequency eddy current. No fatigue cracks or corrosion was detected. The majority of the cracks on the internal structure of the other fuselage sections was detected after 32,000 full pressure cycles. Of these only 25% of the cracks were classified as significant. A number of frame cracks was monitored to obtain crack growth data. The cracks that were classified as not significant were those that did not noticeably grow from detection to 40,000 cycles.

The redesigned internal fuselage structure of the 747-400 nose section was practically crack-free up to 40,000 pressure cycles, verifying the durability of the redesigned structure. Very few cracks were detected in the rest of the fuselage structure. The two significant findings were cracks in the upper deck floor beams adjacent to the stretched upper deck stairwell cutout, and skin cracks in the electrical equipment cooling cutout unique to the -400 model. These cracks were detected past 33,500 test cycles. Production improvements have already been incorporated for these two items, and service bulletins have been issued to address the fleet.

## Pressure Fixture Testing

The 707, designed in the 1950s, was the first commercial jet airliner to be developed at Boeing with a pressurized fuselage. Experience at that time taught the aircraft industry that the ability to tolerate a substantial amount

## 747 Fuselage Fatigue Tests Test Results – Longitudinal Skin Lap Splices (Fay Sealed Configuration – C/L 201 and on)

- -100SR cycled to 40,000 full-pressure cycles; -400 (Sections 41 and 42) cycled to 60,000 full-pressure cycles
- Pressure cycling and teardown inspections have demonstrated that the basic lap splices exceed design goals (20,000 full-pressure cycles)
- Cracks detected in local areas only, beyond 30,000 pressure cycles
- Initial cracking occurs approximately at midbay, well away from frames
- Crack growth data from multiple-site damage scenarios indicates that from initial crack detection to linkup of cracks (crack length after linkup less than one frame bay) is between 10,000 and 15,000 cycles

## Fuselage Damage Tolerance Tests

- Major airframe cyclic tests have been conducted to address damage tolerance

Model	Date	Remarks	Test type
707	1955	Quonset	Discrete source
727	1960	Fuselage pressure test	Fatigue Crack Growth (FCG)
737	1968	Quonset	Discrete source
	1987	Aft fuselage pressure test ▶	FCG
747	1968	Fuselage section	Discrete source, FCG
	1971	Full airframe	FCG
	1990	Fuselage pressure test ▶	FCG
	1990	-300/-400 forward fuselage	FCG
757	1983	Full airframe	FCG
767	1980	Fuselage section	FCG and residual strength
Generic		Fuselage panel in fixture	Starts mid-1990

▶ Retired service airplane



of damage was a requirement of modern airplanes. Therefore, the new pressurized fuselage had to be tested rigorously in order to prove airworthiness. The test structure employed for design development and verification was made of large pressurized panels configured in the shape of a Quonset hut. For the tests, various combinations of skin gages and types of tear straps and shear ties were subjected to saw cuts and punctures by guillotine blades.

To supplement airframe testing, two generic pressure test fixtures were fabricated in 1989. One fixture has a radius of curvature of 74 in (1.9m) to match narrow-body 727, 737, and 757 airplanes. The other has a radius of 127 in (3.2m) to match the widebody 747, 767, and 777 airplanes. Removable test sections are inserted in a cutout in these test fixtures and can be used for fatigue, fatigue crack growth, or residual strength tests. Tests are conducted under pressure loading only, using air as the pressurizing medium. Where appropriate, one crack will be inserted in the test article at a time, and repaired before conducting additional tests. Test sections are modeled using finite element techniques, and analysis results are compared with a comprehensive set of strain gage and crack opening displacement measurements. The structural modeling of crack behavior is refined, as necessary, to provide a validated tool for future airplane design.

## Fuselage Pressure Test Fixtures

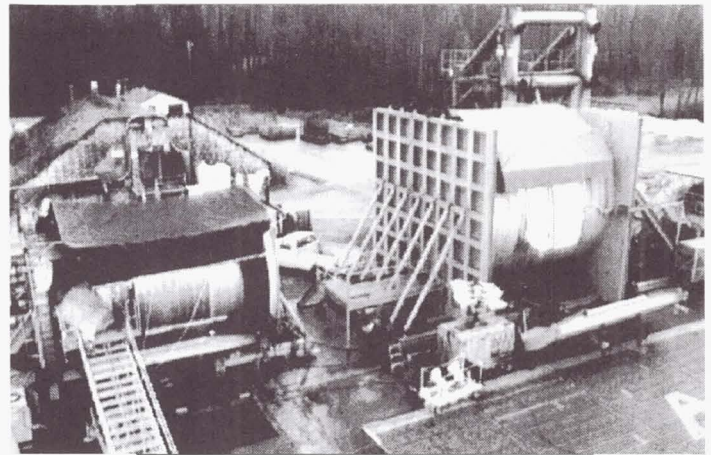
### Objective:

- The test fixtures are used for crack growth and residual strength testing of a variety of fuselage structure configurations:
  - New designs
  - Fleet modifications
  - Repairs
  - Current design
  - Generic studies

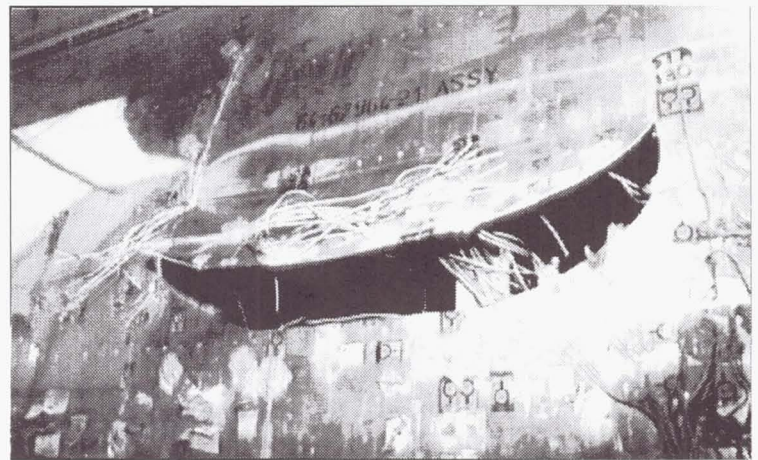
### Status:

- Narrow body tests
  - Total of 8 panels tested
- Wide-body tests
  - Total of 6 panels tested

## Boeing Narrow- and Wide-Body Test Fixtures

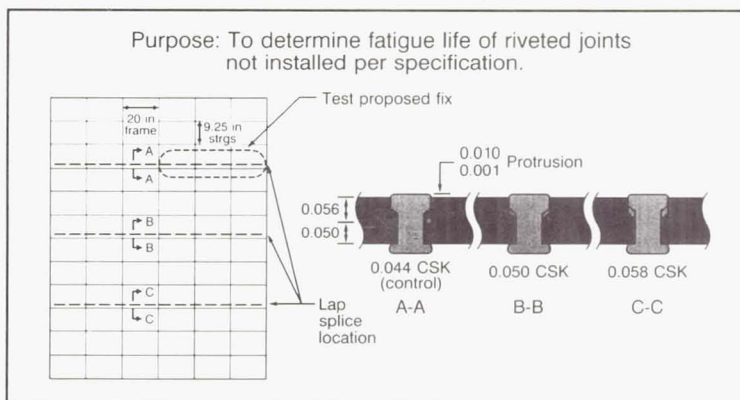


## Boeing Narrow-Body Fuselage Test Panel Bonded Tear Straps Safe Decompression at 30 in



Although major efforts have been focused on the aging fleet, current production models receive substantial attention to ensure continued safe operation with a minimum maintenance burden. One example is the hand driven rivets installed on some 757 lap splices with excessive countersunk depth. Pressure fixture testing demonstrated that these lap joints have sufficient capability to reach lives beyond the SSID inspection thresholds of 37,500 flight cycles. These areas will receive directed inspections at that time to confirm validity of test results.

## 757 Skin Panel Fatigue Test



## 757 Skin Panel Fatigue Test Test Results (200,000 Cycles)

- The fatigue life of a knife edged countersink is greater than indicated by small scale coupon tests
- Crack growth rates have been slower than predicted analytically
- No cracks have been detected in lap splices with 0.050 in deep countersinks or standard countersinks

## TEARDOWN INSPECTIONS

In addition to supporting the inservice fleet, Boeing has instituted a number of other efforts to gain more insight into the present and future behavior of aging structures. These efforts include extensive teardowns and inspection of models 707 and 727 retired from service to determine if any previously unknown or undetectable damage may be starting to develop. These programs started in 1965 and continued through the late 1970s. In some cases, teardown findings have led to design changes, service bulletins, and even airworthiness directives. In general the findings have confirmed previously known or anticipated behavior and have added confidence in both the structures and maintenance programs completed in 1991 as part of industrywide initiatives.

## Boeing Teardown Inspections

- Acquisition of suitable high-flight-cycle airframes
- Disassembly for detailed inspection
 

• 707 wing plus center section	1965
• 707 wing	1968
• 707 wing plus center section and fuselage	1973
• 707 empennage	1978
• 727 forward fuselage	1978
• 737 wing plus center section, forward fuselage, and empennage	1987
• 737 aft fuselage (after fatigue testing)	1988
• 747 wing plus center section and empennage	1988
• 747 fuselage (after fatigue testing)	1991

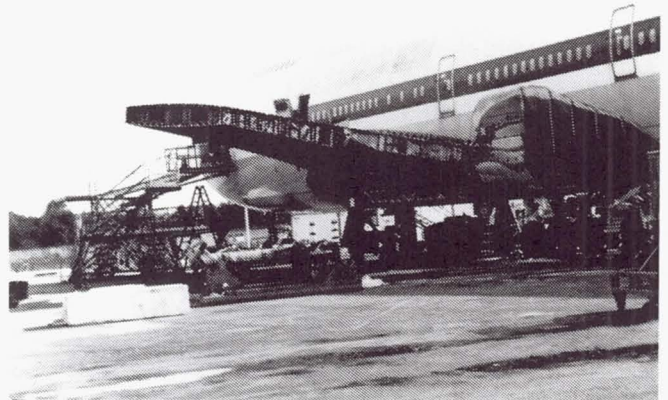


The wings and empennage structure of the retired 747-100 airplane were disassembled for detailed inspections. This airplane, which was retired from service in 1988, had the equivalent of approximately 15,000 long-range flights, which is 75% of the economic design objective. No significant findings requiring fleet actions resulted from the detailed inspections. The fuselage structure of the retired 747-100 airplane was also subjected to teardown inspections after completion of the cyclic pressure testing to 40,000 full pressure cycles. Some internal structure was disassembled and inspected, but the primary focus of the teardown inspections was the skin lap splices. A total of 14,600 upper row fasteners were removed from the lighter gage splices, and high-frequency eddy current inspections of the open holes were conducted. Only six cracks less than 0.05 inch long were confirmed from the teardown inspections. The cracks were in the forward fuselage generally adjacent to cracks detected during pressure cycling. No cracks were detected in the aft fuselage.

The 747-400 forward fuselage test article was also subjected to detailed teardown inspections after completion of the cyclic pressure testing to 60,000 cycles. A total of 8,000 upper row fasteners was removed from the skin lap splices, and high-frequency eddy current inspections of the open holes were conducted. Only 12 cracks less than 0.05 inch long were confirmed from the teardown inspections after 60,000 full pressure cycles. These cracks were also adjacent to cracks detected during pressure cycling. Some internal structure was disassembled and inspected on this test article also, with no significant findings.

One scenario of widespread fatigue damage in skin lap splices that has been put forth is the likelihood of cracks initiating at many fasteners and eventually linking up to form an extremely long crack across the skin panel. While cracks initiated at multiple fasteners on both test articles, evaluation of test results and teardown inspection data demonstrates that the occurrence of the above-stated scenario, due to fatigue, is highly unlikely during the period of cycles covered by the test.

## Wing Removal of Retired 747 Airplane





## FLEET SUPPORT PROGRAMS

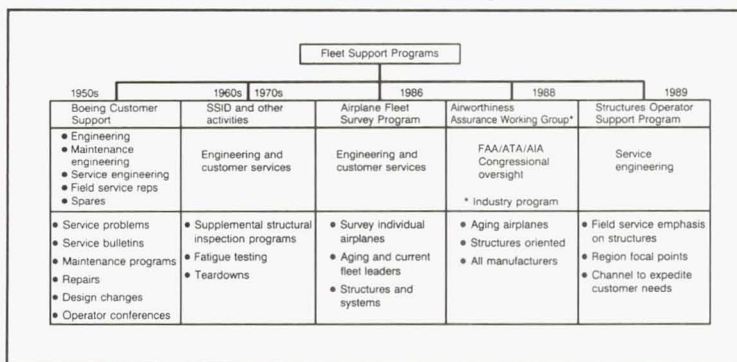
Boeing has in place a number of programs that help support the operation of commercial jet transports and promote a high level of airworthiness in the fleet. Reported service data and other firsthand information from customer airlines are continually reviewed to promote safe and economic operation of the worldwide fleet. Today there are more than 1,000 Boeing airplanes around the world that are over 20 years old, and this number will double by 1995. Approximately 150 Boeing airplanes had reached their design flight cycle goals in 1990, and this number will increase to about 500 by 1995 representing about 10% of the active Boeing fleet.

Inevitably, structural discrepancies are discovered during inspection and maintenance, and well-defined procedures exist to handle these problems wherever in the world they occur. In the United States, they are reported to the manufacturer and/or the FAA. The manufacturer in turn will inform all operators to determine if other operators have experienced similar problems. In many cases, the manufacturer will make a design change in production and/or issue a service bulletin that describes the problem and background and suggests means of corrective action for aircraft in service. For the majority of problems there is no safety issue and therefore no requirement for the operator to follow the service bulletin recommendations. However, for economic reasons, many operators will follow them, either by inspection and/or terminating modification action.

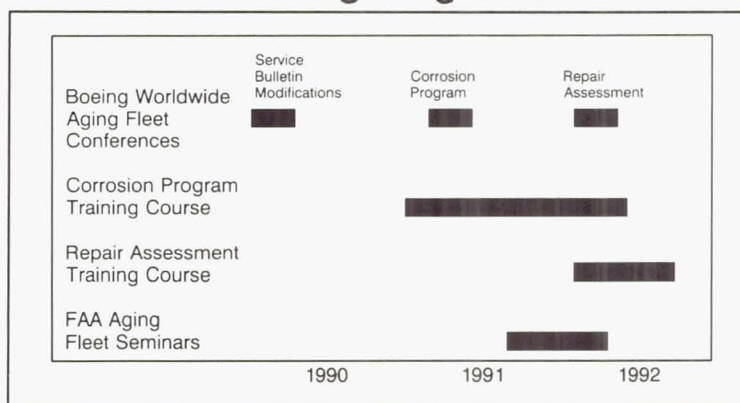
Structural maintenance is a cornerstone of continuing safety in commercial air transport. Aging fleet concerns have focused unprecedented attention on additional requirements, and the industry and airworthiness authorities are implementing additional maintenance actions to ensure continued safety. These are focused on mandatory modification rather than continued inspection, development of improved mandatory corrosion prevention and control programs, consolidation of basic maintenance programs, updates of supplemental structural inspection programs, and development of guidelines to determine the adequacy of structural repairs.

A major part of successful implementation of these programs is to provide adequate training of operator

## Fleet Support Programs



## Aging Airplane Awareness and Training Programs



## Corrosion Prevention and Control Program Training

- Training program developed to achieve consistent implementation
- All groups involved with the program were invited -
  - Airline operators
  - Leasing companies
  - Repair stations
  - Regulatory agencies
  - Etc.
- The FAA also participated as instructors for the airworthiness directives section of the training program
- The course has been given 19 times in Seattle and 17 times in regional locations
- Over 1,300 people from nearly 300 groups have attended the training course

maintenance staffs. A number of courses addressing both general guidelines as well as specific corrosion prevention and control program details have been conducted worldwide by Boeing.

The lessons learned from the 707/727/737/747 corrosion prevention and control programs have also resulted in specific recommendations for update of the Maintenance Planning Documents (MPD) through the Maintenance Review Board (MRB) process. Similar activities are also progressing for models 757/767.

Recognizing this trend, Boeing implemented a new program to better understand the condition of older airplanes. The Aging Fleet Survey Program was developed during 1986 and implemented in 1987. The success of the program resulted in expansion to include new production airplane models such as the 737-300, 757, and 767, some of which have already seen 8 years of service. By reviewing the structural and systems performance of these airplanes during their early and maturing years, it is believed that Boeing and the operators can take the right actions to preclude the majority of aging fleet problems 10 years or so from now.

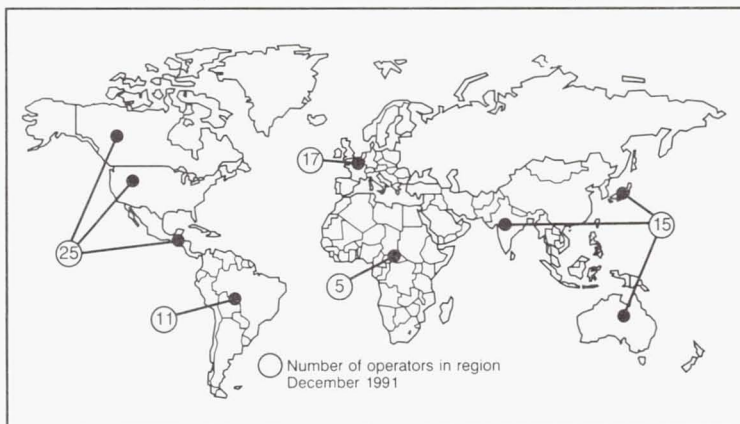
Airline acceptance of the program and cooperation with Boeing has been positive. Observing a significant number of airplanes in a variety of operating and climatic environments around the world provides a composite sample of each model and a better understanding of common and unique aspects within and between model fleets. As of December 1991, 138 airplanes owned by 73 operators have been visited in 40 countries around the world. Although the number of airplanes observed is a small percentage of the total fleet, it does represent about 15% of the high-time airplanes.

Considerable variation has been observed in both airplane condition and airline maintenance procedures, such as inspection intervals and corrosion prevention measures. The condition of the structure was generally good but considerably below expectation in a few cases. All significant findings pertaining to a specific visit are recorded and assigned to appropriate organizations for necessary fleet action. The collected data have been pooled for fleet evaluations to determine if there are trends requiring additional actions by Boeing and/or the

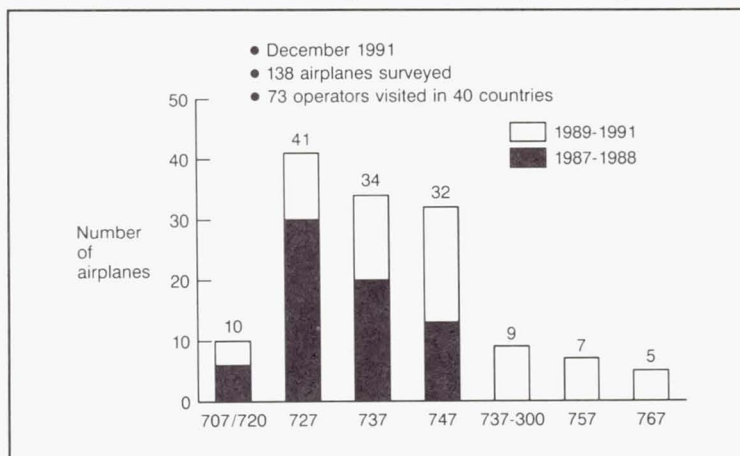
## 757/767 Corrosion Prevention and Control Program Background

- A corrosion prevention and control program was proposed by Boeing for the 757/767 and endorsed by the AATF (AAWG)
- Structures working groups were formed and 757/767 corrosion programs were developed similar to the 727/737/747 programs
- 757/767 corrosion program was presented to and accepted by the industry steering committee on September 11, 1991
- Documentation and FAA control provided through MRB and MPD

## Boeing Fleet Survey Participants



## Airplanes Surveyed During Program





operators. A number of detailed action items have resulted from the surveys, and their applicability has been reviewed across all Boeing models. To ensure anonymity, all identifiable operator/airplane data are treated as confidential.

The fleet survey program is continuing with 26 airplanes observed in 1991. Recent findings indicate the condition of the airplanes generally is good and that they are receiving adequate and conscientious maintenance. Most discrepancies noted by the Boeing teams had already been recorded by the operator and corrective action was under way. Recent surveys have confirmed our belief that the aviation industry is more aware of the necessity of applying timely corrosion control and installing well-engineered repairs and, generally, has a more heightened awareness relative to sound maintenance practices.

## DESIGN IMPROVEMENT PROCESS

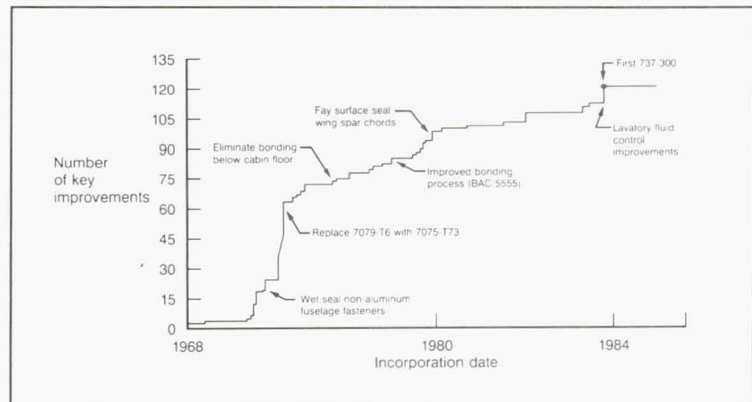
As airplanes mature, service experience provides a natural focus on design improvements, which are reflected in the relative modification costs for aging airplanes as they reach their design service objectives. A better understanding of how airplanes withstand the rigors of long-term use will help future generations of commercial airplanes to be safer and less maintenance-intensive with age. Just as today's airplanes have benefited from previous lessons learned, so must the knowledge and experience gained from today's efforts be used to improve the quality and performance of future airplanes. There will certainly be even more emphasis during design and construction on reducing the potential for corrosion. Specific design goals include improved corrosion-resistant alloys and finishes, improved sealing and drainage, and increased attention to accessibility and inspectability. These goals rank equally with strength, damage tolerance, durability, and cost/weight efficiency.

Boeing uses the experience of its 5000-airplane fleet to incorporate over 140 corrosion resistance improvements developed for new and derivative models. New aluminum material tempers are selected to provide high resistance to stress corrosion and galvanic corrosion. Graphite composites are isolated to inhibit galvanic

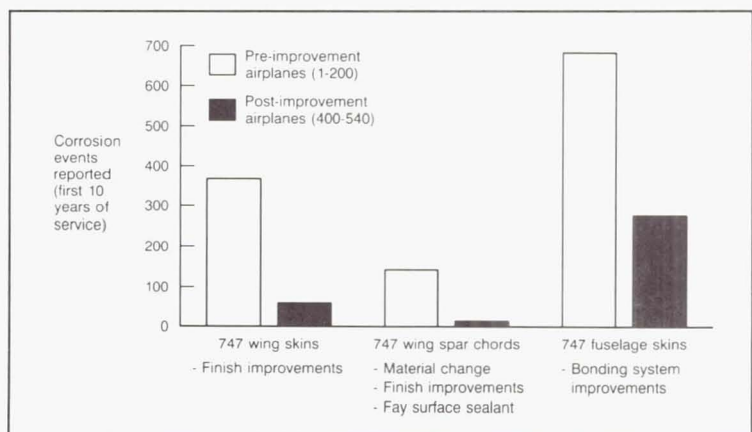
## Airplane Fleet Survey Program Findings on 757/767

- A few, minor fatigue cracks only
- No stress corrosion cracking
- Some areas of corrosion
  - Generally, evidence shows corrosion improvements are effective
  - Areas under lavatories and galleys still pose problems
  - Drain paths not kept free of debris

## Typical Corrosion Improvements Implemented on Boeing Airplanes



## Effect of Corrosion Improvements on Model 747



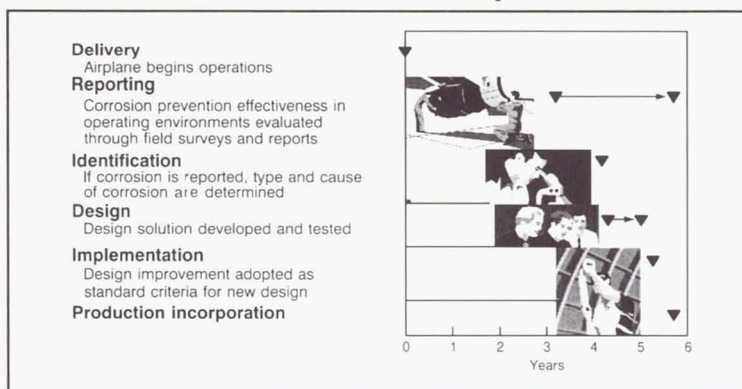


actions. No alloy steel fasteners are used in exterior skins. High-durability adhesive bonding with phosphoric-acid anodize surface preparation and corrosion-inhibiting adhesive primer are used to eliminate disbonding. This bonding durability has been proved by over 20 years of research and 14 years of in-service airline experience. Fuselage drainage has been improved by new, flexible leveling compound filler materials that eliminate wet pockets. Faying surface seal is used on wing details that attach to exterior surfaces of the skin. One-piece module construction with a water-tight base is used to reduce corrosion in lavatories on newer models.

The development phasing of many of these corrosion protection improvements depends on timely inspection reporting to identify problems and to take necessary design improvement steps. The time typically required spans a 5- to 6-year cycle to achieve full implementation of corrosion prevention measures.

Boeing has recently developed a comprehensive Corrosion Design Handbook reflecting fleet experience to provide the structural engineer with the same corrosion prevention expertise that parallels methods used to develop producible, durable, and damage-tolerant structures. Similarly, improved structural arrangements and concepts will enhance the inherent robustness and forgiveness of the structure, facilitate simpler repairs when damage occurs, and facilitate accessibility and inspectability. The knowledge gained in the past 4 years will also enable better focus on the initial and continuing maintenance needs of new airplanes. In turn, this will allow the most effective and timely distribution of airline resources to maintain their airplanes indefinitely to achieve continued safe and economic operation.

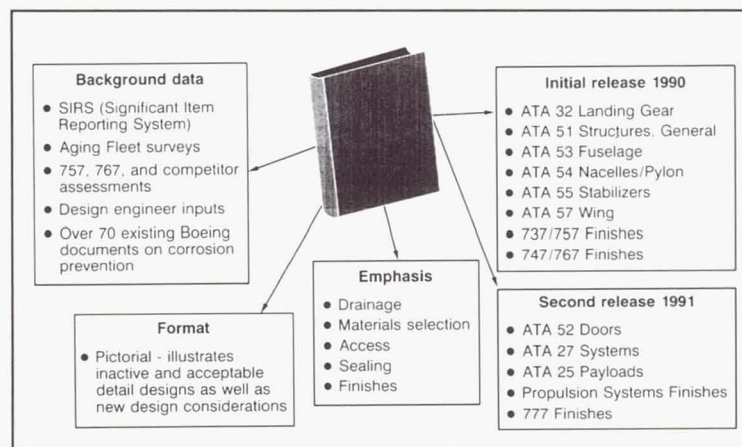
## Corrosion Prevention Improvements



## Structural Damage Technology Handbooks



## Design for Corrosion Prevention Handbook



Knowledge from 30 years of producing and supporting jet airliners has been applied to the new wide body design. The fuselage shell, for example, employs new structural features to improve producibility, maintainability, and inspectability. Stiffeners with a hat cross section have been replaced by Zee stiffeners, which have fewer surfaces to attach or join. A circular cross section has simplified the frames, which can be roll-formed. Lessons from the older airplanes in service have also affected the design. The outer skin has been made thicker in proportion and drainage has been improved by eliminating discontinuities that are the result of bonded edges, tearstraps, and machined pockets.

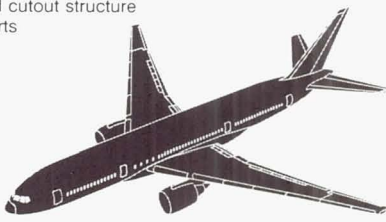
## 777 Fuselage Design Body Monocoque

Improved producibility, maintainability, and inspectability

- Z versus hat stiffeners
- Simpler frames
- Simplified cutout structure
- Fewer parts

Incorporated "Aging Fleet" lessons learned



- Made outer skin more "robust"
- Improved drainage
- Controlled fluid flow



## WIDESPREAD DAMAGE

Original fail-safe design criteria required limit-load strength for a fatigue failure or obvious partial failure of a single principal element. The capability to analyze damaged structure has improved in recent years through the application of advanced fracture mechanics. Federal regulations cited in FAR 25 were updated in 1978 to reflect such state-of-the-art developments to require consideration of damage growth at multiple sites in damage tolerance assessments. These new criteria also require attention to structural damage characteristics when developing an inspection program. This has accelerated development of better ways to ensure damage tolerance of new and aging jet transports by timely detection of unexpected fatigue, environmental deterioration, or accidental damage.

## Damage Tolerance Regulations

Analysis	FAR 25.571 (pre-1978)	FAR 25.571 (post-1978)
Residual strength	<ul style="list-style-type: none"> <li>• Single element or obvious partial failure</li> </ul> 	<ul style="list-style-type: none"> <li>• Multiple active cracks</li> </ul> 
Crack growth	<ul style="list-style-type: none"> <li>• No analysis required</li> </ul>	<ul style="list-style-type: none"> <li>• Extensive analysis required</li> </ul>
Inspection program	<ul style="list-style-type: none"> <li>• Based on service history</li> <li>• FAA air carrier approval</li> </ul>	<ul style="list-style-type: none"> <li>• Related to structural damage characteristics and past service history</li> <li>• Initial FAA engineering and air carrier approval</li> </ul>

The general understanding of widespread structural damage (WSD) is often linked to the idea of an aircraft suffering damage over large regions of its structure. However, it is the synergistic effect of these damages which is identified as the principal concern. In this case, the approach for determination of the crack growth and the residual strength of isolated damages are no longer sufficient for determination of the inspection intervals. Therefore, the approach for maintaining damage tolerance of the structure needs to be reconsidered for both aging airplanes and for new design.

Those types of multiple site damage and multiple element damage whose extents are within the existing damage tolerance assumptions are termed as "local."



## Widespread Fatigue Damage Definitions:

The presence of fatigue cracks at multiple sites of the airplane structure such that the interaction of these cracks degrades the damage tolerance capability of the structure more than any one crack acting independently.

There are two types of widespread fatigue damage:

- **Multiple Site Damage (MSD)**

Simultaneous development of fatigue cracks at multiple sites in the same structural element, such that fatigue cracks may coalesce to form one large crack.

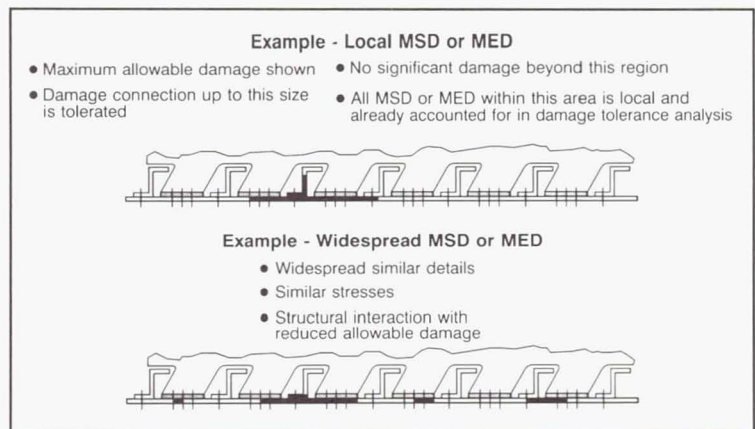
The initial multiple site damages could develop in one element from a number of initiation points, which are operating at similar stresses, are of similar design, and/or are in one cross section. These initial cracks are independent, and usually non-uniform, but could result in a significant increase in the crack propagation rate in the concerned element, compared to the propagation of any single crack, and may result in a reduction of the airframe damage tolerance capability.

- **Multiple Element Damage (MED)**

Simultaneous development of fatigue cracks in similar adjacent structural elements, leading to an interaction of these cracks.

The initial fatigue cracks would typically develop simultaneously in similar adjacent elements of a multiple load path arrangement. These fatigue cracks are initially independent, and usually non-uniform, but may become dependent as they grow. This could result in a significant increase in the crack propagation rate in the concerned elements and/or a reduction of the residual strength capability.

## Example – Local Versus Widespread MSD or MED



## Structures Potentially Susceptible to Widespread Fatigue Damage

### Fuselage

- Longitudinal skin joints, frames, and tear straps
- Circumferential joints and stringers
- Frames
- Aft-pressure dome outer ring and dome-web splices
- Other pressure bulkheads attachment to skin and web attachment to stiffener
- Stringer-to-frame attachments
- Window surrounding structure
- Overwing fuselage attachments
- Latches and hinges of nonplug doors
- Skin at runout of large doubler

### Wing and empennage

- Chordwise splices
- Rib-to-stiffener attachments
- Skins at sheer tie locations
- Stiffener modifications at tank-end ribs



## SUMMARY

Boeing is dedicated to design and manufacture of safe commercial jet transports. The successful accomplishment of this responsibility over the last three decades has contributed significantly to a position of industry leadership and reflects the top priority given to safety. The structural integrity assurance of commercial airplane structures is a serious and disciplined process. High standards must be maintained to ensure the safety of aging airplanes until economics dictate their retirement. Standard Boeing practices to ensure continuing structural integrity include providing structural maintenance programs, continuous communication through customer support services, and recommendations for maintenance actions through service letters, structural item interim advisories, and service bulletins. To help identify potential problems associated with the aging jet transport fleet, Boeing has implemented additional activities:

- Fatigue testing of older airframes to determine structural behavior in the presence of service-induced problems such as corrosion and repairs.
- Teardown of older airframes to help identify corrosion and other structural service defects.
- An engineering assessment of the condition of a representative sample of older Boeing airplanes to observe effectiveness of corrosion prevention features and acquire additional data that might improve maintenance recommendations to the operators.

The design, construction, operation, and maintenance of airplanes take place in a changing and dynamic arena, with new technology, new needs, and new players. The structural safety system may never be perfect but has produced an enviable record, and the aging fleet initiatives will measurably improve that record. If the lessons being learned today by the manufacturers, the operators, and the authorities are properly reflected in our next-generation airplanes, they should fly longer and more safely with progressive maintenance that ensures continued structural airworthiness until retirement from service for economic reasons.

- Ongoing programs to support aging aircraft fleet
- Similar activities on current production models to prevent future aging fleet issues
- Lessons learned incorporated in new designs



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## MAINTAINING THE SAFETY OF AN AGING FLEET OF AIRCRAFT

A.W. Hoggard\*

The current status of the Douglas Aging Fleet is examined in light of increasing concern for the possibility of the onset of widespread cracking and recent industry activity to minimize the concern. A fleet monitoring program together with an augmented maintenance program is proposed as a possible means to reduce the concern. Six candidate options for maintenance program augmentation are examined which have been shown to be effective in detection of widespread fatigue damage. A brief example of how this system might be applied to the DC-9 Fleet is presented.

### INTRODUCTION

The prospect of aircraft being operated for extended periods of time is clearly upon us. It seems that modern technology and ingenuity has kept our modern jet fleet intact and economical to operate far beyond what was envisioned at the time the aircraft was first certified. Well over 2500 aircraft, worldwide, are beyond their original 20 year design life goal. But are these aircraft structurally safe for such operations? Or will there be an increase in the number of accidents like the one in April of 1988, in which a portion of the forward fuselage of an aircraft separated due to undetected widespread structural degradation? This paper presents one manufacturer's viewpoint on the means to maintain the safety of an aging fleet of aircraft against such structural degradation.

### DOUGLAS AGING FLEET

Figure 1 shows the current statistics for the Douglas fleet of jet aircraft. A total of five different models are in service around the world. As of July 1, 1991 nearly 2900 aircraft have been produced, of which over 2500 are currently active, 175 have been retired, and 139 have been destroyed. Of the active aircraft, 1170 have exceeded their original design life goals in terms of flight hours, landings and/or length of service. DC-8 Fuselage Number 13 has currently accrued 32 years of service. This represents service life greater than 1.5 times the original estimate of 20 years. In fact the average aircraft in the DC-8 and DC-9 fleet has exceeded the 20 year goal originally established. DC-9 Fuselage Number 12 has exceeded 93,000 landings, well over 2.25 times the original design goal of 40,000 landings. DC-10 Fuselage Number 117 has exceeded 77,000 flight hours which represents nearly 1.3 times the original design life goal of 60,000 flight hours.

Of the 2500 active aircraft, not one has reached its test supported life. This life equals the tested landing life divided by a scatter factor of two and is generally considered to be the life beyond which the fatigue behavior of the fleet is not proven. Aircraft operated up to this point should be relatively free from widespread fatigue cracking.

Aircraft operated beyond the test supported landing life require a greater level of surveillance in order to protect them against the potential effects of several forms of widespread fatigue damage. This concept is shown in Figure 2 for the DC-9 Fleet. Based on statistical inference, the projected life of the fleet is

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\* MANAGER, AIRWORTHINESS ASSURANCE PROGRAMS,  
DOUGLAS AIRCRAFT COMPANY



considerably beyond the test supported life. This gives rather strong evidence that the DC-9, barring any unforeseen economic or regulatory issues, will be in use for some time to come.

## ECONOMIC ISSUES

The cost of operation has historically been the deciding factor in when to retire an aircraft. In recent times the cost of fuel has moderated. This, coupled with the availability of a whole host of performance improvement items such as re-engining programs, the addition of winglets and electronic engine performance monitors has made the older aircraft as or more economical to operate as acquiring new generation aircraft. Barring any unforeseen down turns in the economy, there is no evidence that aircraft will be retired in the near future for economic reasons.

## INDUSTRY AND REGULATORY INITIATIVES

Damage tolerance requirements were introduced to the industry in 1978. Figure 3 presents a time line of significant regulatory and industry initiatives for structural design since the advent of these requirements. More recently, following the April 1988 accident, operators, manufacturers, and regulators began developing a series of programs to reduce the future probability of a similar event. These are also reflected in Figure 3.

Industry initiatives. In August of 1988, the Air Transport Association (ATA) and Aerospace Industries Association (AIA) in cooperation with the Federal Aviation Administration (FAA) established the Airworthiness Assurance Task Force (AATF). The AATF was commissioned to evaluate the deficiencies in the system that lead up to the April 1988 accident and to propose both industry wide and model specific actions to fix the system. To date the AATF has instituted programs dealing with:

1. Mandatory modifications for termination of repetitive inspections for certain critical service bulletins.
2. Industry-wide mandatory corrosion prevention and control program.
3. An industry guide on proper maintenance programs for aging aircraft as well as model specific guidance.
4. A periodic audit of the Supplemental Inspection Documents(SID).

Additionally, the AATF is currently working on proposals dealing with repairs to aircraft and improved industry communication.

The AATF programs provide a means to upgrade an aircraft at a specific point in its life to insure that the the original fail safe design principles are intact on an aircraft by aircraft basis. This insures that the first line of defense against all forms of structural degradation is still effective. These programs are only marginally effective, however, in the prediction or prevention of the onset of widespread fatigue damage.

Regulatory initiatives. In recent years the FAA has proposed rule making in two areas which may limit the useful life of the older aircraft.

In response to the April 1988 accident, the FAA independently proposed a series of new rules to further reduce the chances of having another similar accident. The proposed rules embrace a fatigue testing concept to protect the fleet against the onset of widespread fatigue damage and are directed towards new certification programs as well as retroactive fatigue testing for aircraft already certified.

The AATF as well as the AIA have initiated independent reviews of the proposed rules to assist the FAA in arriving at appropriate rule making material. Preliminary results of the industry reviews indicate that while fatigue testing is appropriate for new certification programs it is not appropriate, in and of itself, to protect older aircraft against the onset of widespread fatigue damage. Specific rules requiring fatigue testing of older aircraft may mean the premature and unnecessary retirement of many aircraft.

In addition to the safety related proposed rules, the FAA has also recently proposed new noise related rules which will accelerate the phase out of noisier Stage 2 aircraft by the year 2000. The enactment of these rules may have a significant effect on reducing the size of the aging fleet in the near term. Figure 4 shows the comparison between the current scheduled phase out and the new proposed rule.

Although economics and regulatory issues may have an impact, many first and second generation jet aircraft will be in extended operation for some time to come. In operating these aircraft, there are concerns, both perceived and factual, about structural integrity that need to be addressed. Those concerns center around the timely detection of widespread fatigue damage.

### DAC AGING AIRCRAFT PROGRAMS

Douglas, with the cooperation of operators and regulators throughout the world, has developed five interrelated programs to identify and address issues that are generic to an aging fleet of aircraft. These programs, are shown in Figure 5.

Some of these programs are passive in that they act as a source of data to monitor the fleet. These programs include the Extended Fatigue Test Programs and the Douglas Design Evaluation Program. The data from these programs are used to adjust, as necessary, active programs. Active programs include the Supplemental Inspection Document program and the AATF initiatives discussed earlier. These programs are active in the sense that they perform specific tasks on aircraft in the fleet. Training, which is interactive, is also included amongst the programs because there is, as always, a need to insure a common level of understanding about the programs across the industry.

All of these programs establish a basis which supports the timely detection of all forms of structural degradation. Figure 6 depicts the interaction that is needed between the the various parts of the industry to insure the structural integrity of the aircraft. Each party has its own specific role in assuring that integrity. The manufacturer supplies the necessary technical support in evaluating service findings. The operator is ultimately responsible for the maintenance of the safety of the aircraft. And the regulator insures that the regulations are being followed by both the manufacturer and the operator and insures compliance with critical safety issues. The key ingredient to this system is communications between all of the parties.

### WIDESPREAD FATIGUE DAMAGE

Widespread fatigue damage is best understood by looking at a structural life concept that embodies damage tolerance principals as shown in Figure 7. Damage tolerance principals often assume the presence of discrepancies in structure to facilitate the analysis procedure. These discrepancies are known as equivalent flaw sizes and in fact account for a number of variables in the manufacturing and assembly process.

As aircraft are being manufactured, quality control procedures insure that the finished products are essentially free from structural discrepancies. Discrepancies that do slip by the quality control procedures are generally sub-detectable and random in nature. The larger of these discrepancies (some times known as a rogue flaw), if located in a critical area, could ultimately become a fatigue crack and grow to a critical size at some point in the aircraft life. In the design of a new aircraft, the crack growth design life goal for the rogue flaw is normally set at a life greater than the design life goal of the aircraft. The program described by the Supplemental Inspection Document is designed to find and correct such flaws before they reach critical size.

The smaller discrepancies, known as initial quality flaws, initiate fatigue cracks and grow at a much slower rate but are much more frequent in occurrence. This leads to the idea that there are many discrepancies at many locations growing simultaneously which may ultimately lead to a failure condition. These smaller discrepancies are more likely to be a concern on configurations that are highly repetitive, such as the rows of rivet holes in a high load transfer splice. In the most extreme case, structural failure occurs by net section yield along a line of small cracks that are only marginally detectable. This describes the most classic form of widespread fatigue damage. Fatigue test of a full scale component to at least two design



lifetimes followed by a teardown inspection is normally used to rule out this form of structural degradation for newly certified aircraft.

There are two specific forms of widespread fatigue damage that are of potential concern to the aging fleet of aircraft. The first is known as Multiple Site Fatigue Damage (MSD) and the other is known as Multiple Element Fatigue Damage (MED) (see Figure 8).

#### MULTIPLE SITE FATIGUE DAMAGE (MSD)

MSD is fatigue damage that principally occurs in highly repetitive design details such as rivet holes in splices and is generally limited to one principal structural element such as a wing or fuselage skin panel.

#### MULTIPLE ELEMENT FATIGUE DAMAGE (MED)

MED is fatigue damage that occurs in multiple stiffening members that support the skin panels causing the fail-safe properties of the design to be downgraded through loss of strength in similar adjacent stiffening elements.

#### DESIGN ISSUES

Careful design supported by testing can assure an aircraft has a long life free from widespread cracking. Proper attention to material selection and detail design standards for areas of load transfer with uniform high stress levels can mean a design that is essentially free from MSD or MED for its intended life.

The DC-9, currently in its third lifetime of operation, is a good example of this. In 1981 Douglas repurchased DC-9 Fuselage Number 3 for extended fatigue testing to insure an adequate fatigue margin of safety for extended operations. At the time, it was one of the five highest time aircraft having accumulated over 66,500 landings. It was in regular airline service and was accruing landings at the rate of 3500 landings a year. It was scheduled for a major structural maintenance check (D check) within a month. After the aircraft was flown to Douglas Aircraft Company in Long Beach California, it was stripped of all interior items and a thorough inspection of the structure conducted. After a total of 15 years of service, the structure was found to be remarkably free of corrosion and fatigue cracks. The problem areas that were found were either repaired or tagged for careful evaluation as the test progressed. Certain service bulletins were selected for incorporation before the test began.

The program extended the tested life on the DC-9 fuselage structure from 125,000 flights to over 208,000 flights. At the end of the test an extensive evaluation of the structure was conducted including the 'tear down' of over twenty-two feet of fuselage longitudinal lap splice. Over 8000 individual rivet holes were checked for fatigue cracks with no findings reported. In addition, after fifteen years of service and five years of testing, there was no reported corrosion found in the faying surface. Additional coupon type fatigue tests on the wing and empennage structure extended the tested life of these components to over 208,000 flights.

#### MAINTENANCE ISSUES

When an aircraft enters service it is the responsibility of the operator to perform routine maintenance checks in order to detect and correct damage or deficiencies in a timely manner. The requirements of most regulators stipulate that an acceptable maintenance program must be in place before the airline initiates service with the aircraft. At the time of certification the manufacturer, with help from the customer airlines and the regulator, publishes an initial maintenance program which can be used as an acceptable guide to the development of individual airline maintenance programs. The structural maintenance stipulated in this initial program is based on visual detection of structural degradation. This concept did not come about by chance but is based on the fail-safe design principals required by regulators and insisted upon by operators. Figure 9 shows an example of what is meant by visual detection of structural degradation. In this particular



example, any cracks developing in the exterior skin at rows B or C are acceptable. Cracks that develop in any other manner are unacceptable.

DAC assures visually detectable modes of failure in aircraft design details through a combination of coupon, sub-component, and full scale fatigue tests. As a note of interest the DC-10 full scale fatigue test was delayed for a number of months because of a hidden failure that developed in one of the splice joints in a component test. As a result of the failure, a new design was developed and tested in another component test to verify its failure mode prior to incorporation of the design modification into the full scale fatigue test.

The primary issue here is that widespread fatigue cracking may not be visually detectable before failure (see Figure 7). Because of this, a competent program must be in place to insure the timely detection of widespread damage as the aircraft age.

### AN INDUSTRY APPROACH TO DETECTING WIDESPREAD FATIGUE DAMAGE

As part of the original Airline/Manufacturer Recommendations (1) following the April 1988 accident it was recommended that the industry "Continue to pursue the concept of teardown of the oldest airline airplane to determine structural condition, and conduct fatigue tests of older airplanes.....". In June of 1990 the AATF commissioned an International Task Group of operators and manufacturers to investigate and propose appropriate actions based on this recommendation. The AATF further stipulated that the investigation should place special emphasis on the ability of the actions proposed to discover widespread damage in a timely manner.

The Task Group, composed of six manufacturers and five operators, worked closely with regulators from four countries to develop appropriate actions for the aging fleet. The Task Group reached seven major conclusions and six recommendations during its eight month study and published a 190 page report.\* The primary conclusion reached was that while significant improvements in the structural safety system have been introduced by the AATF programs, there is still an outstanding concern for the potential onset and possible non-detection of widespread fatigue damage. The report further went on to recommend that it is appropriate to perform a model specific audit of those aircraft that have exceeded or are about to exceed their original design life goals with respect to the programs already in place to determine if any concerns exist that require additional actions. This audit would be monitored by an Industry Oversight Group to insure that audit of each model is consistent with industry norms.

### THE MODEL SPECIFIC AUDIT

In consideration of the potential consequences of widespread fatigue damage, it is important to assess the likelihood of its occurrence on a given airplane and the ability of the current maintenance system (as modified by the SID, manufacturer and AATF programs) to discover the damage in a timely (safe) manner. In order that this is done in a logical and consistent manner across the industry, the Task Group proposed the audit process, shown in Figure 10, be applied to each candidate fleet of airplanes after full definition of all the aging aircraft related activities. This process would be conducted in the AATF Structures Working Group environment with operators, the airplane manufacturer and the regulators fully involved. Specific courses of actions developed by the Structures Working Group would be integrated into the aircraft maintenance program through a Service Bulletin, SID update, or changes to the model specific aging aircraft maintenance guidelines.

### CANDIDATE MODEL SPECIFIC ACTIONS

The Task Group performed an extensive data collection and analysis activity to determine candidate options that have applicability to the identified concerns. This activity centered around the past manufacturer, operator and regulator experience with the related areas.

The activities of the Task Group formed the basis for the development of alternative options to resolve any residual concern for the potential of widespread fatigue damage in a fleet of aircraft. The effort also

\* ATA Report, 'A Report of the AATF on Fatigue Testing and/or Teardown Issues', Unpublished - please contact the Air Transport Association, Washington DC, for a copy, February 1991



served to provide some common ground to assess the benefits and limitations of each option as well as the ground rules under which each option should be considered.

All of the options adopted are valid to one degree or another in predicting the onset and location of multiple site damage and multiple element damage, but none of the options are foolproof. The potential consequences of multiple site and multiple element damage dictate the need for safeguards that will provide a high degree of confidence that it will not happen. Ultimately this will depend on conscientious and reliable inspections of the aircraft structure.

A total of six options were identified as possible candidates to be used to alleviate any residual concerns. These options can be viewed singly or in combination to develop the required level of safety. The options identified are shown in Figure 11 and discussed below.

Selected limited non-destructive disassembly and refurbishment of high time airplanes continuing in service. This option would allow a manufacturer to identify specific areas of an airplane that may be susceptible to multiple site damage or multiple element damage for an increased level of surveillance during scheduled maintenance visits. Candidate airlines and airplanes would be targeted for partial disassembly in the areas identified to facilitate inspections for emerging structural degradation. This process would allow smaller damage levels to be found than would be found by non-disassembly techniques. This option has distinct advantages over other options because of lead time and resources required to implement the process. In general, it is believed that this option would be able to supply the largest volume of the highest credibility data for assessment of structural condition.

Continuing assessment of the fleet demonstrated capability through diligent monitoring of service experience. This option has been successfully used in the past to predict future behavior of the fleet by statistical inference. The process examines the characteristic shape of a histogram representing the fleet of aircraft. Given a statistical theorem (e.g. Weibull or Log-Normal), it is possible to project the mean life of the fleet and then determine the proven life based on some predetermined criteria (either one crack in the fleet or sometimes a predetermined failure rate). This process, used with other options discussed here, can become a powerful tool in predicting the onset of widespread cracking.

Fleet exploration of high time airplanes with improved and state of the art NDI techniques. This option is clearly something that is focussed for the future. The viability of this option lies with the research and development initiatives of the FAA and NASA and to a certain extent on developments in the human factors area. It is included here because it is expected that the developments in the NDI area will be evolutionary rather than revolutionary. The process of evaluation should always be ready to accept innovative concepts as they are introduced especially if they save time and increase accuracy.

Testing of new or used structure on a smaller scale than full component tests, i.e. sub-component and/or panel tests. This option represents another method to evaluate the inherent capabilities of a particular design as far as susceptibility to multiple site damage and multiple element damage is concerned. This option allows a rapid assessment of various repetitive design details at a nominal cost in resources.

Extended full scale fatigue test and teardown. This option requires a significant investment in resources and time but does provide significant insight to future fatigue related behavior of the airplane design including modes of failure and probable locations and would allow adjustment of maintenance programs accordingly. There are concerns about how representative a single test of a particular airplane is to the whole fleet, and the amount of time required to complete the testing, teardown and release the results to the operators. There is also a significant concern associated with current popular connotation of fatigue testing setting a safe life on the airframe.

Airplane teardown. This option allows opportunistic destructive teardown inspections of airplanes, that for some reason, have been removed from service. This option allows a detailed piece by piece examination of each component of an airframe to discover structural degradation and establish structural condition. The process can identify suspect areas and project future structural performance. Fractographic analysis of the

structure, post teardown, can be used to enhance the value of the teardown inspection. There is a significant concern of how representative the tear down of a particular airplane is of the whole fleet. The option requires a considerable investment in resources but can be completed in a reasonable time frame.

The Task Group Report (2) has been officially transmitted to the FAA and they are currently considering the merits of the report's conclusions and recommendations.

### MONITORING THE FLEET

The key element in the implementation of the model specific audit requirements is the monitoring of fleet findings in a timely and expeditious manner. Figure 12 shows how the fleet findings might be monitored in a flow chart form. The Structures Working Group would propose model specific audit requirements. These audit requirements would then be implemented either by augmenting the SID programs, issuance of service bulletins or the manufacturer would start a fatigue test. Operators owning designated aircraft would then be required to modify their maintenance programs in accordance with the recommendations of the Structures Working Group as endorsed by the Regulator. Findings would be complied as a result of the inspections and acted upon when warranted. This procedure is not unlike the current procedure used for the Douglas Aircraft Company SID programs.

### MAINTAINING SAFETY

If the key element in the audit program is the monitoring of fleet findings, the second most important element is the interpretation of the findings. In the formulation of the audit program there must be a conscientious evaluation of the ability of a particular action to identify problems in terms of what is already known about the fleet of aircraft. In reality this is a modification of the fleet demonstrated life approach (Item B) shown in Figure 11.

At Douglas Aircraft Company, the accepted convention for maintaining safety for potential problems is a probability of 90 percent detection before there is possibility of exceeding a Failure Rate (FR) of  $1 \times 10^{-9}$  on a per flight basis. The probability of not exceeding the failure rate is the basis of selecting the appropriate candidate option and the frequency of application considering routine maintenance and the current SID. It is important to point out that these areas have no known history of fatigue cracking and findings as few as one can initiate the fleet protection program.

An example of how this system might work for the DC-9 fleet is shown in Figure 13. Each design detail/area under consideration has its own estimated mean fatigue life ( $N_{\text{ainst}}$ ) based on fatigue tests or other comparable means. Using a failure rate as a basis, a life ( $N^{\text{TH}}$ ) can be calculated using the statistical equation:

$$FR = \frac{p(N^{\text{TH}+1}) - p(N^{\text{TH}})}{1.0 - p(N^{\text{TH}})} \quad (1)$$

Where  $p(N^{\text{TH}})$  is the log-normal probability of having a crack  $a \geq a_{\text{inst}}$  at  $N^{\text{TH}}$  based on the estimate of mean life and the standard deviation (s). This failure rate represents only the possibility of having a crack of critical length. This value must be multiplied by the probability of encountering limit load to arrive at the  $1 \times 10^{-9}$  joint probability.

Using the estimate of mean life, the standard deviation and the current fleet statistics, it is possible to estimate the total population of cracks in the fleet for a given design detail. As shown, the life represented by  $N^{\text{TH}}$  is somewhat in advance of the lead aircraft yet there is a distinct probability of the presence of 2.3 cracks in the active DC-9 Fleet, if the estimate of mean life is correct. Fleet safety is assured if there is a 90%



probability that the maintenance programs will detect the emerging cracks in the candidate fleet (shown shaded in Figure 13) before the lead aircraft reaches  $N^{TH}$ . Negative findings (e.g. inspections that find no evidence of fatigue cracking) provide the basis for extending the useful life of the aircraft. Positive findings, on the other hand, provide the basis for a fleet protection program.

### CONCLUSIONS

Barring prohibitive economic or regulatory issues, the fleet of aging aircraft will continue in service for many years to come. As these aircraft age there is an increased likelihood of widespread fatigue damage. The concerns associated with the possibilities of the onset of widespread damage require competent programs to insure that this form of structural degradation is discovered in a timely manner. A model specific audit can identify areas that have a potential for future development of widespread damage. An augmented maintenance program similar to the program offered in the Supplemental Inspection Document can then be formulated on an area by area basis using one or more of the following options:

- A. Selected limited non-destructive disassembly and refurbishment of high time airplanes continuing in service.
- B. Continuing assessment of the fleet demonstrated capability through diligent monitoring of service experience.
- C. Fleet exploration of high time airplanes with improved and state of the art NDI techniques.
- D. Testing of new or used structure on a smaller scale than full component tests, i.e. sub-component and/or panel tests.
- E. Extended full scale fatigue test and teardown.
- F. Airplane teardown.

Inspection conducted in the fleet under the augmented maintenance program can then be compared to the expected fleet behavior. Results of this comparison will provide the necessary information to increase the life of the fleet or begin a fleet protection/modification program.

### ACKNOWLEDGEMENTS

The author wishes to thank Joan Hughson, Group Leader, Aging Aircraft Programs, Douglas Product Support for her invaluable assistance in helping write this paper.

## SYMBOLS USED

AATF	=	Airworthiness Assurance Task Force
AIA	=	Aerospace Industries Association
ATA	=	Air Transport Association
$a_{\text{detectable}}$ or $a_{\text{det}}$	=	Detectable crack (or flaw) size associated with a 90% probability of detection, for a given Nondestructive method
$a_{\text{instability}}$ or $a_{\text{inst}}$	=	Crack size at onset of rapid propagation, corresponding to 100% limit load.
$a_{\text{manufacturing}}$	=	Initial quality flaw size attributable to the manufacturing process.
$a_{\text{net section yield}}$	=	Crack size at onset of rapid propagation due to net section yield.
FAA	=	Federal Aviation Administration
FR	=	Per flight failure rate based on the joint probability of having a crack $a \geq a_{\text{inst}}$ and encountering 100% limit load.
MED	=	Multiple Element Fatigue Damage
MSD	=	Multiple Site Fatigue Damage
$N^{\text{TH}}$	=	Fatigue life threshold based on a given failure rate.
$N_{\text{ainst}}$	=	Mean life for a crack to grow to $a_{\text{inst}}$ (includes crack initiation life)
NDI	=	Nondestructive inspection (includes visual)
$p(N^{\text{TH}})$	=	Probability of having a crack $a \geq a_{\text{inst}}$ during the $N^{\text{TH}}$ th flight
$p(N^{\text{TH}+1})$	=	Probability of having a crack $a \geq a_{\text{inst}}$ during the $N^{\text{TH}} + 1$ flight
SID	=	Supplemental Inspection Document
s	=	Standard deviation of (log N) normal distribution

## REFERENCES

- (1) DOT-TSC-FA890-88-26, 'Proceedings of THE INTER-NATIONAL CONFERENCE ON AGING AIRPLANES', Appendix B, August 1988



JULY 1, 1991

99,581,314 FLIGHT HOURS  
71,390,011 LANDINGS

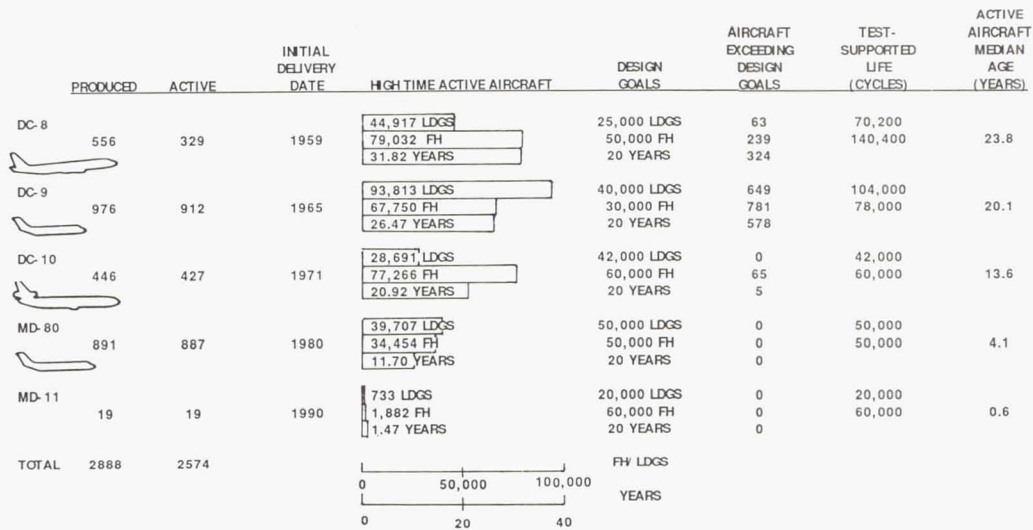


Figure 1 Douglas fleet statistics - July 1, 1991

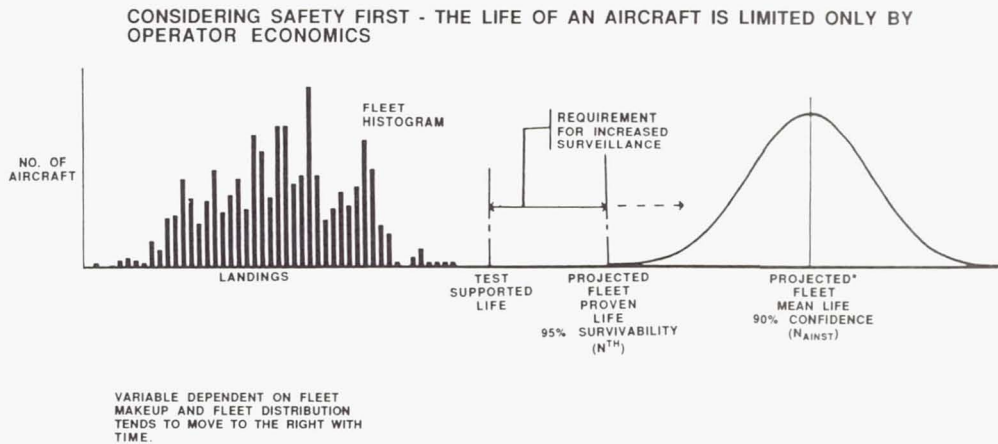


Figure 2 Fleet projected/proven life

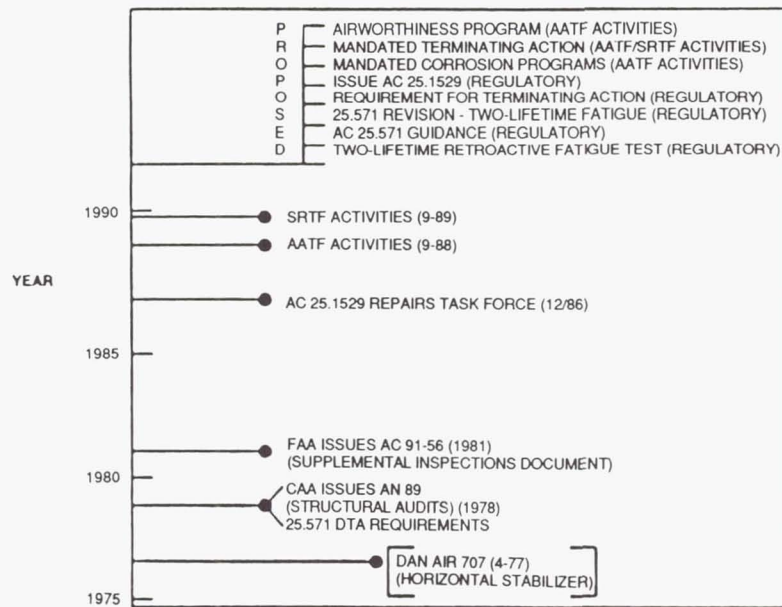
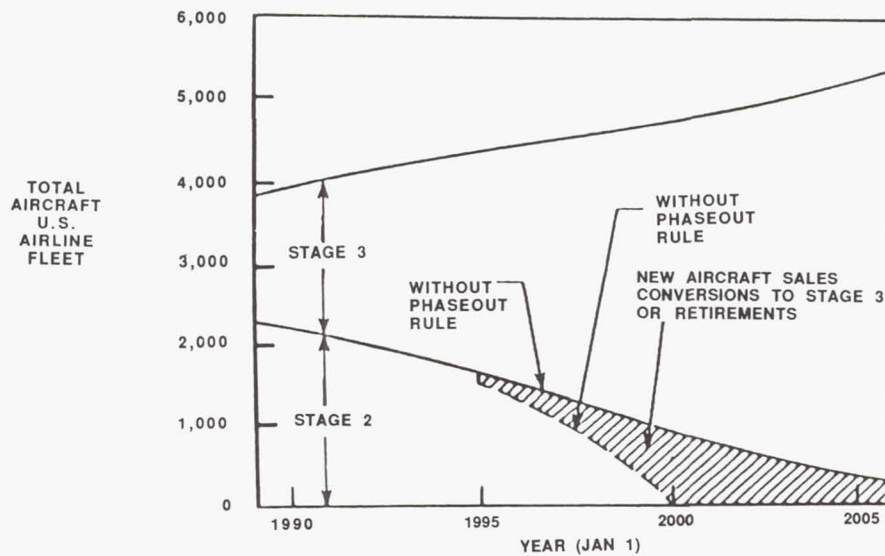
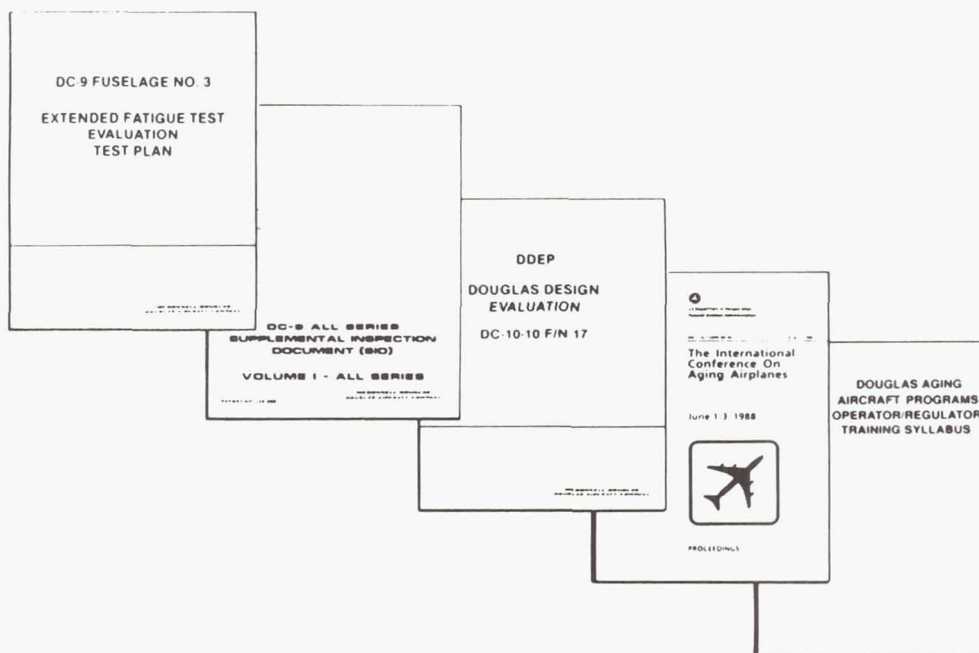


Figure 3 Airframe regulatory issues



### MAY EFFECT NUMBER OF OLDER AIRCRAFT

Figure 4 Stage 3 noise - proposed requirements



## AUGMENTS ROUTINE MAINTENANCE PROGRAMS

Figure 5 DAC aging aircraft programs

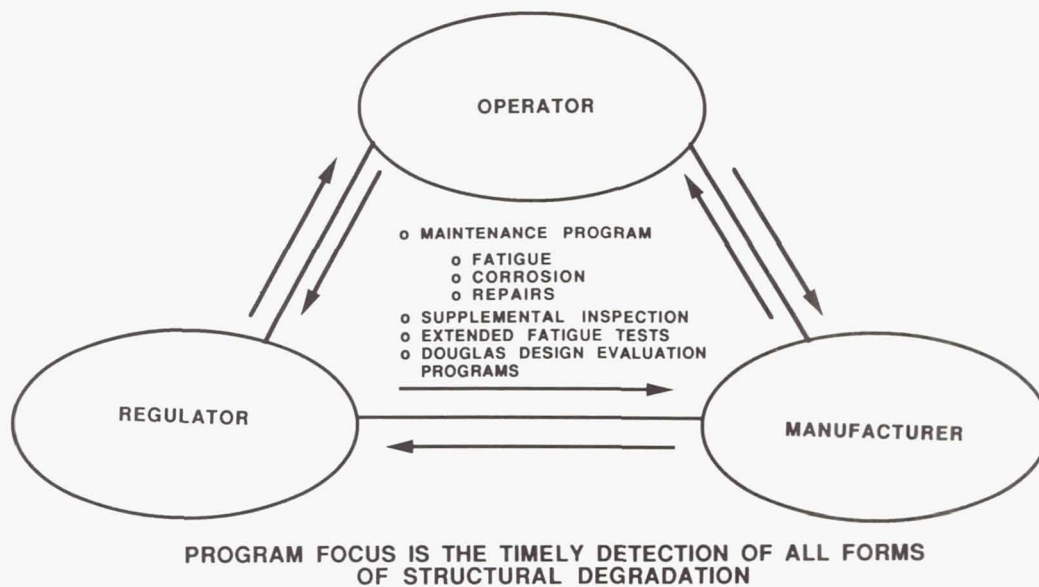


Figure 6 Maintaining the safety of the aging fleet



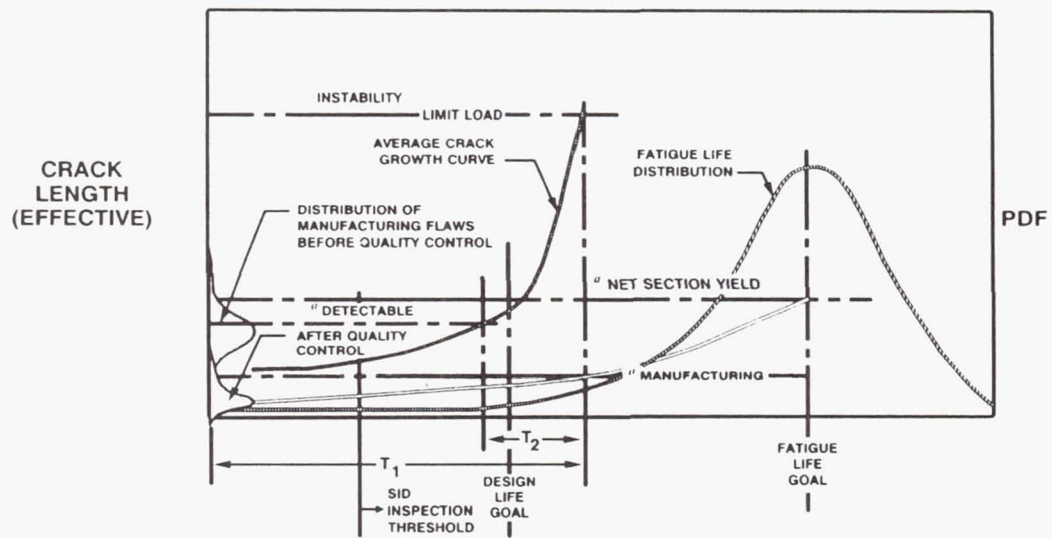


Figure 7 A structural life concept

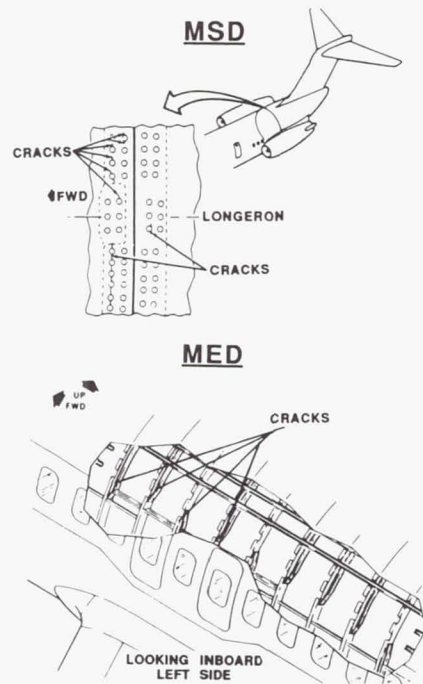
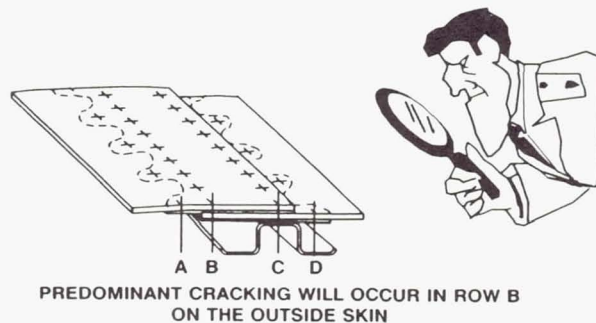


Figure 8 Multiple site and multiple element fatigue damage

o INITIAL MAINTENANCE PROGRAM BASED ON VISUAL INSPECTION

SPLICE INSPECTIBILITY



o MSD AND MED MAY NOT BE VISUALLY DETECTABLE

THEREFORE, COMPETENT PROGRAM MUST BE IN PLACE

Figure 9 Maintenance program issues

1. DETERMINE AREAS POTENTIALLY SUSCEPTIBLE TO MSD.
2. DETERMINE AREAS OF POSSIBLE CONCERN FOR MED.
3. ASSESS EACH SUSPECT AREA'S LEVEL OF SAFETY WITH CURRENT AND AUGMENTED MAINTENANCE PROGRAMS.
4. SELECT AREAS REQUIRING ADDITIONAL MONITORING TO ESTABLISH THE REQUIRED LEVEL OF SAFETY.
5. DETERMINE ADDITIONAL AREA SPECIFIC ACTIONS TO ACHIEVE THE REQUIRED LEVEL OF SAFETY.
6. IMPLEMENT APPROPRIATE ACTIONS.

Figure 10 Model specific audit for MSD/MED

- A. SELECTED LIMITED NON-DESTRUCTIVE DISASSEMBLY, INSPECTION AND REFURBISHMENT OF HIGH TIME AIRPLANES CONTINUING IN SERVICE.
- B. CONTINUING ASSESSMENT OF THE FLEET DEMONSTRATED CAPABILITY THROUGH DILIGENT MONITORING OF SERVICE EXPERIENCE.
- C. FLEET EXPLORATION OF HIGH TIME AIRPLANES WITH IMPROVED STATE OF THE ART NDI TECHNIQUES.
- D. TESTING OF NEW OR USED STRUCTURE ON A SMALLER SCALE THAN FULL COMPONENT TESTS (I.E. SUB-COMPONENT AND/OR PANEL TESTS).
- E. FATIGUE TEST OF HIGH TIME AIRPLANE OR FULL SCALE MAJOR COMPONENT FOLLOWED BY DETAILED TEARDOWN OF TEST ARTICLE.
- F. TEARDOWN OF HIGH TIME AIRPLANE.

Figure 11 Model specific candidate actions

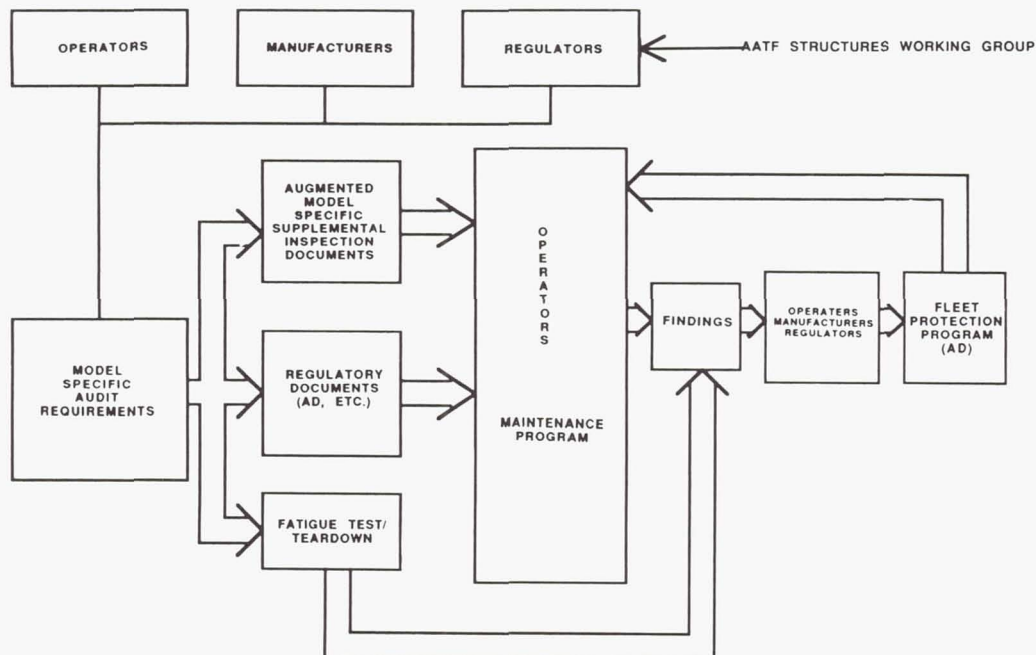


Figure 12 Monitoring of fleet findings



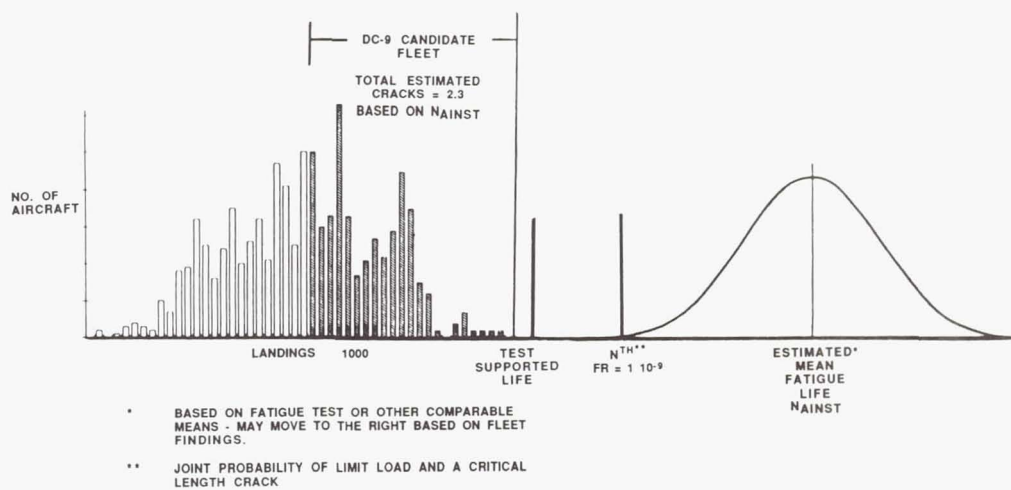


Figure 13 Maintaining the desired level of safety

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# PERFORMANCE OF FUSELAGE PRESSURE STRUCTURE

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## ABSTRACT

There are currently more than 1,000 Boeing airplanes around the world over 20 years old. That number is expected to double by the year 1995. With these statistics comes the reality that structural airworthiness will be in the forefront of aviation issues well into the next century.

This paper describes the results of previous and recent test programs Boeing has implemented to study the structural performance of older airplanes relative to pressurized fuselage sections. Included in testing were flat panels with multiple site damage (MSD), a full-scale 737 and 2 747s as well as panels representing a 737, a 777, and a generic aircraft in large pressure-test fixtures.

Because damage is a normal part of aging, this paper focuses on the degree to which structural integrity is maintained after failure or partial failure of any structural element, including multiple site damage (MSD), and multiple element damage (MED).

## BACKGROUND

The 707, designed in the 1950s, was the first commercial jet airliner to be developed at Boeing with a pressurized fuselage. Experience at that time taught the aircraft industry that the ability to tolerate a substantial amount of damage was a requirement of modern airplanes. Therefore, the new pressurized fuselage had to be tested rigorously in order to prove airworthiness.

The test structure employed for design development and verification was made of large pressurized panels configured in the shape of a Quonset hut. For the tests, various combinations of skin gages and types of tear straps and shear ties were subjected to saw cuts and punctures by guillotine blades. Some early combinations proved inadequate as they resulted in dynamic destruction of the test panels. Eventually, several light-weight designs were verified to safely tolerate a high degree of damage (Figure 1).

At that time, MSD was not included in the Quonset testing, but was present in the full-scale fatigue tests of a production configuration.

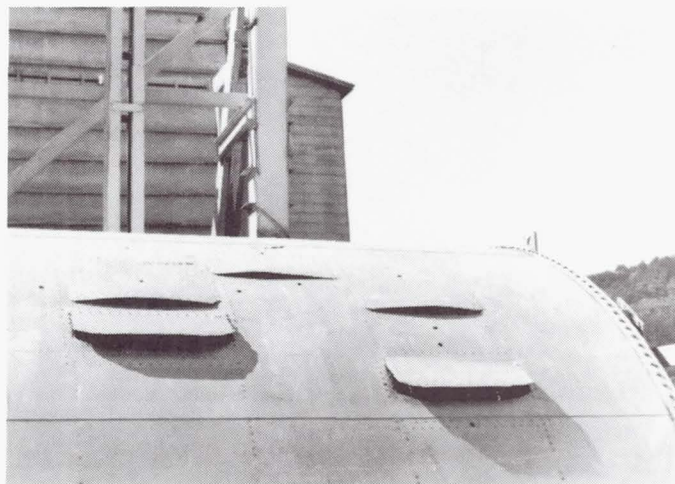


Figure 1. 707 Test Panel.

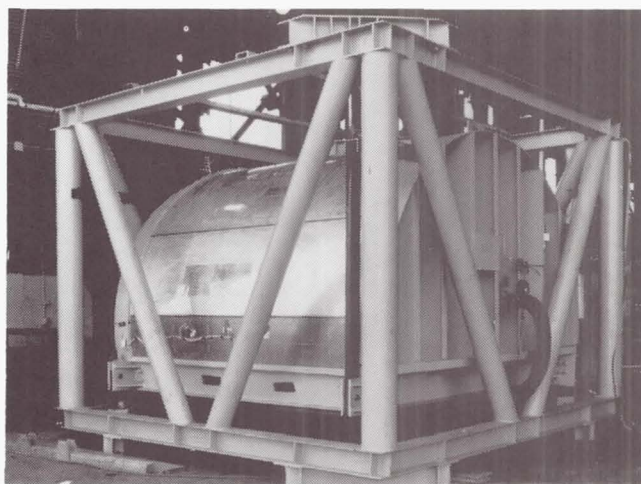


Figure 2. 737 Test Panel.

During full-scale fatigue tests, cracks developed in lap splices with many cracks at fasteners in adjacent bays. These were natural fatigue cracks (MSD) that grew until they eventually linked up in one bay, then formed flaps that allowed safe decompression.

Structural designs that incorporate tear straps and/or shear ties under the fuselage frames at 20-inch spacing were verified by guillotine and fatigue tests and subsequently also verified in service by the 707, 727, 747, 757, and 767.



Special tests were devised for the 737, which was a lighter weight. Because of lower operating cabin pressure and lower body bending loads due to its shorter fuselage, the 737 was designed with a thinner 0.036-gage skin.

Again, a Quonset hut test panel was used for development, except the test fixture for the 737 was loaded with skin shear in addition to internal pressure (Figure 2).

The final design for the 737 incorporated a waffle doubler bonded to the 0.036-gage skin with tear straps at 10-inch intervals.

Dual 12-inch-wide guillotine blades straddling a frame were used to test its structural damage tolerance (Figures 3 and 4). The results of these tests are shown in Figure 5.

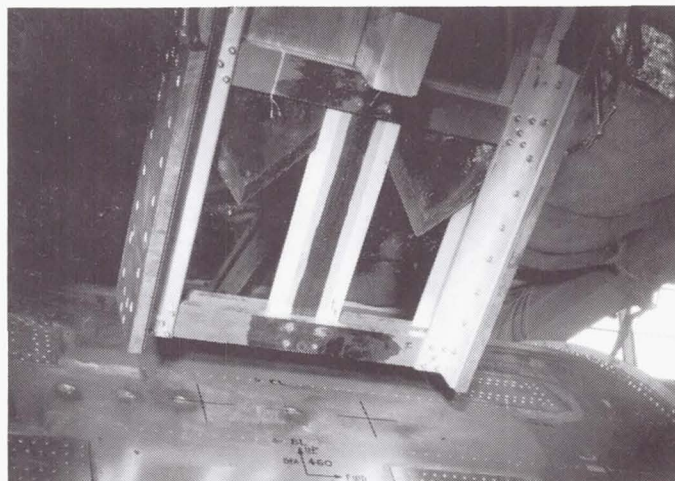


Figure 3. Double Guillotine Blade Test.

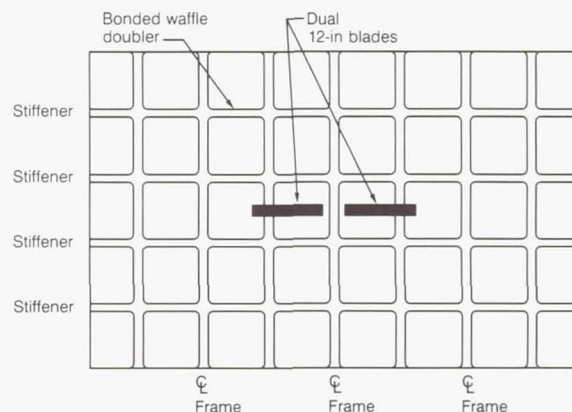


Figure 4. 737 Developmental Guillotine Test Panels.

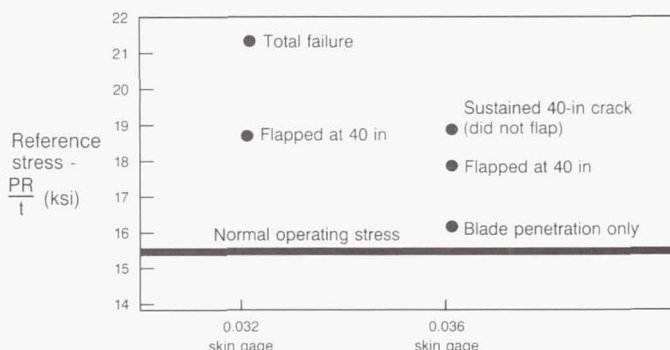


Figure 5. 737 Developmental Guillotine Test Results.



Figure 6. 737 Guillotine Test.

The tests demonstrated the capability of the 737 pressure structure to safely tolerate a substantial amount of damage, frequently cracks up to 40 inches long (Figures 6 and 7).

Saw cut and guillotine tests were also conducted on the 747 design. The use of tear straps and shear-tie designs created a fail-safe capability for cracks of at least one bay, or 20 inches long.

The primary difference noted between these tests and previous 707 and 737 tests was the tendency of the heavier 0.063-gage skin not to form flaps.

The designs of the 757 and 767 fuselage sections were essentially an improved version of the 707. During verification on a full-scale fuselage and after fatigue testing to twice the design objective, the designs demonstrated the ability to sustain two-bay cracks with a severed middle frame. Lighter skin gages, i.e., 0.040, on the 757 allowed safe decompression by flapping.

The design of the 727 fuselage section was so similar to the 707 design that test data were applicable to both and no additional testing was necessary.

### MULTIPLE SITE DAMAGE CONSIDERATIONS

In the 1980s, with many airplanes in the world fleet approaching or exceeding their original design service objectives, the focus of the industry turned to damage scenarios associated with fatigue cracking.

Tests using saw cuts and guillotine blade punctures demonstrated a high level of damage containment whether the source of the damage was accidental or due to local fatigue. However, widespread MSD fatigue cracking at many locations raised new questions.

Boeing had relied on data produced by 707 fatigue tests that demonstrated safe decompression in lap splices when natural cracking occurred in several adjacent bays, but they were uncertain what would happen if the natural MSD was more severe than had occurred due to fatigue scatter. Also, more information was needed for heavier skin gages that tended not to form flaps.

To address these concerns, Boeing initiated several test programs:

- Flat panel tests using flat panels, rather than the more elaborate Quonset hut, to obtain an early indication of the probable effects on residual strength of a common scenario—small fatigue cracks remaining in a row of fastener holes after some cracks have linked up to become a large crack.
- Fatigue tests on a 737 aft fuselage section from an aircraft previously in service.
- Fatigue tests on a 747-100SR fuselage section from an aircraft previously in service.
- Fatigue tests on a new 747-400 forward fuselage.
- Construction of two large fuselage pressure-test fixtures and testing of a variety of 737, 777, and generic test panels.

New data obtained from these test programs were examined to determine the effects of MSD on the damage tolerance performance of fuselage pressure structure.

### FLAT PANEL TESTING

Natural fatigue cracking with MSD frequently involves small cracks at fastener holes in a row preceding a larger crack in the same row. Flat panels were used in this test program as a quick method for testing residual strength, while avoiding the time of loading curved, more realistic fuselage panels. For this reason, data obtained by this method were regarded as approximations only. The following three test locations were selected for flat panel testing (Figure 8):

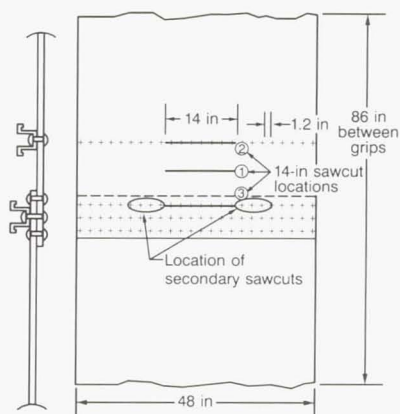


Figure 8. Flat Panel Test Specimen.

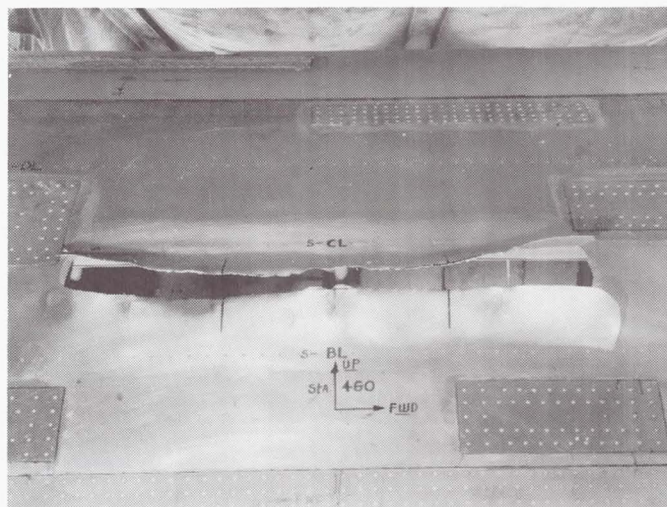


Figure 7. Example of Safe Two-Bay Flapping.

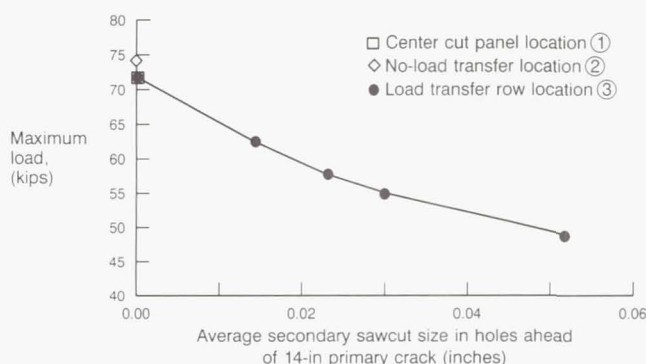


Figure 9. Flat Panel Test Results.



Location 1: A baseline for testing material properties involving a saw cut, but without a row of holes preceding the cut.

Location 2: A row of holes in an area where no load is transferred to test for the effect of holes without MSD.

Location 3: The top row of a lap splice to include the effects of both fastener holes and load transfer; also used to determine the effect of small fatigue cracks, simulated by saw cuts approximately 0.02 inches wide, on the residual strength of the panel containing a large crack.

Test results are shown in Figure 9. The graph indicates that there was no significant difference in residual strength between locations 1, 2, and 3 until the small MSD saw cuts were added to location 3. At that point, the amount of load tolerated rapidly declined as the size of the small saw cuts increased. Because of this significant drop, the phenomena will be investigated further under more realistic conditions using a pressure-loaded test fixture.

### 737 AFT FUSELAGE FATIGUE TEST

Another step in evaluating the effect of MSD on damage tolerance was the fatigue testing of a retired 737 aft fuselage (Figure 10). After 59,000 service flights, the fuselage test section (Figure 11) was cycled until normal fatigue cracking had begun and grown to its natural conclusion of a two-bay crack with safe decompression by flapping (Figure 12).



Figure 10. Retired 737 Aft Fuselage Fatigue Test Article.

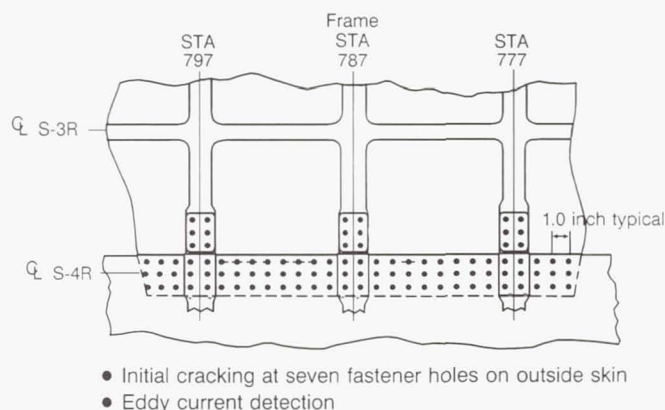


Figure 11. Initial 737 Lap Splice Cracking.

In this test, MSD was present in adjacent holes and in adjacent frame bays with a nonuniform distribution of crack sizes typical of fatigue scatter. Although the test may not represent the worst-case of fatigue scatter and MSD, it is reasonable to assume that the test results do represent typical performance.

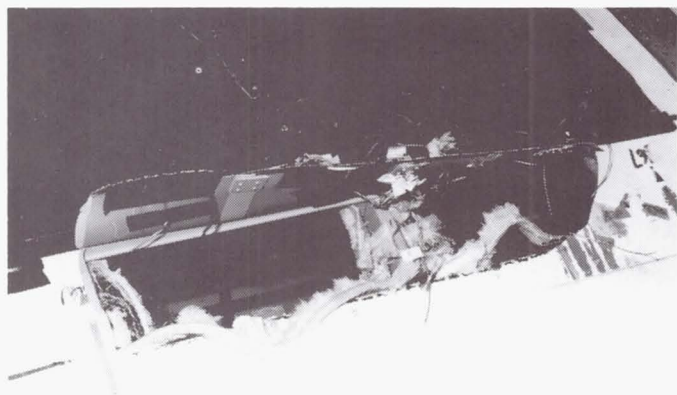


Figure 12. Controlled Decompression of 737 Fuselage Test Article.

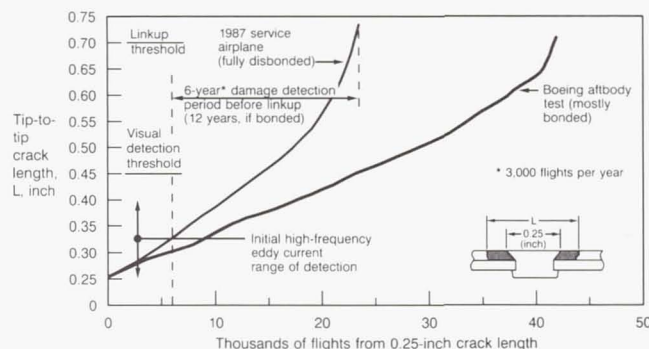


Figure 13. 737 Lap Splice Crack Growth Data.



Decompression by flapping is not relied on as a safety factor in the case of cracks in lap splices. Rather, inspection programs are in effect that ensure crack detections before linkup or very soon thereafter. In the test described above, a 12-year damage detection period between initial detection and linkup was indicated assuming 3,000 flights per year (Figure 13). According to further experience on a fully disbonded inservice airplane, that number would be reduced to 6 years—still ample time for detection.

### 747-100SR FATIGUE TEST

For this test, a 747-100SR with an equivalent of 20,000 full-pressure cycles (flights) was obtained from service and monitored as it was extended to 40,000 cycles (Figure 14). The minimum gage lap splices are shown in Figure 15.



Figure 14. Retired and New Model 747 Fuselage Test Articles.

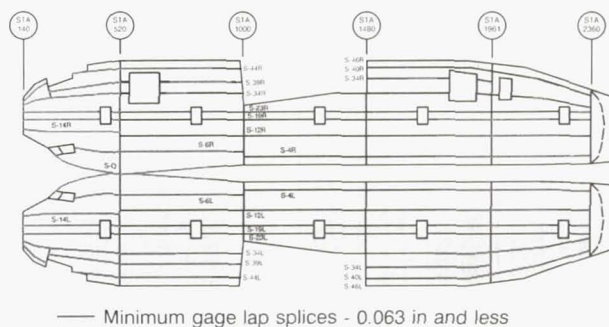


Figure 15. 747-100SR Fuselage Fatigue Tests.

An initial crack was detected in the lap splice in S-14R at 21,500 cycles. This crack eventually linked up with other small cracks and grew to approximately 6 inches by the time the test was concluded at 40,000 cycles (Figure 16).

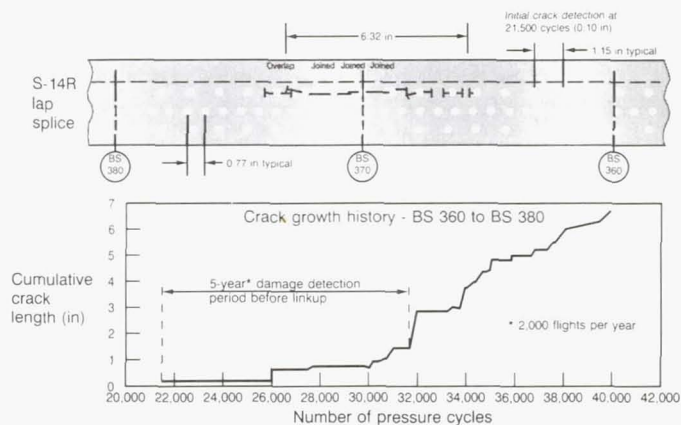


Figure 16. 747-100SR Fuselage Fatigue Test Results.

Assuming 2,000 flights per year of normal operations, the crack growth data from this test indicate a 5-year damage detection period before linkup. After linkup, tests indicate there is a significant, additional safe damage detection period.

### 747-400 FATIGUE TEST

Sections 41 and 42 from the 747-400 production line were pressure cycled to determine some of the fatigue and damage tolerance characteristics of the latest production configurations (Figure 14). The minimum gage lap splices are shown in Figure 17.

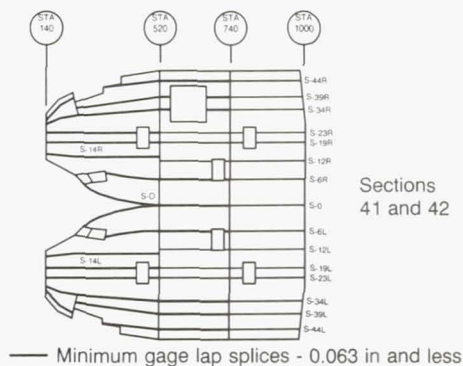


Figure 17. 747-400 Fuselage Fatigue Tests.

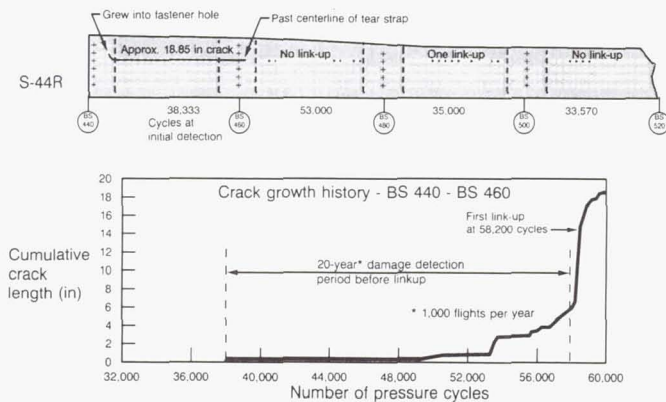


Figure 18. 747-400 Fuselage Fatigue Test Results—S44R

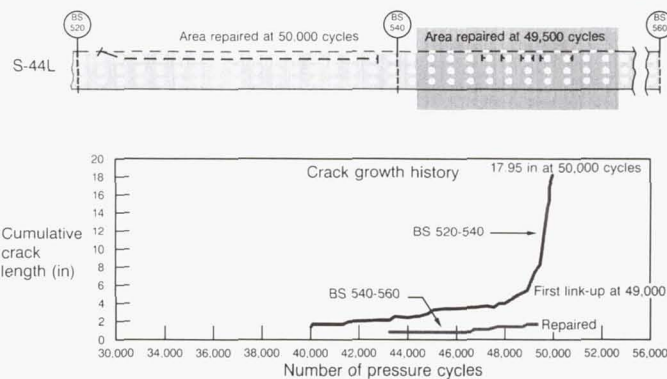


Figure 19. 747-400 Fuselage Fatigue Test Results—S44L

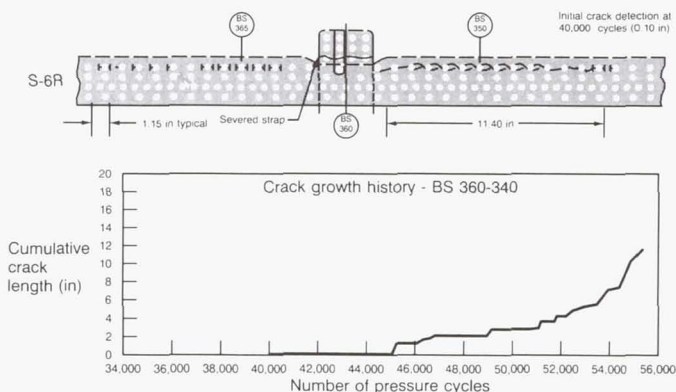


Figure 20. 747-400 Fuselage Fatigue Test Results—S6R.

accommodate testing for fatigue, crack growth, and residual strength of large pressure panels with a variety of structural designs and details.

A typical test panel configuration is shown in Figure 22; however, up to three lap splices can be accommodated and test locations can vary. Test panel frame spacing, stringer spacing, and panel radius are set by the fixture.

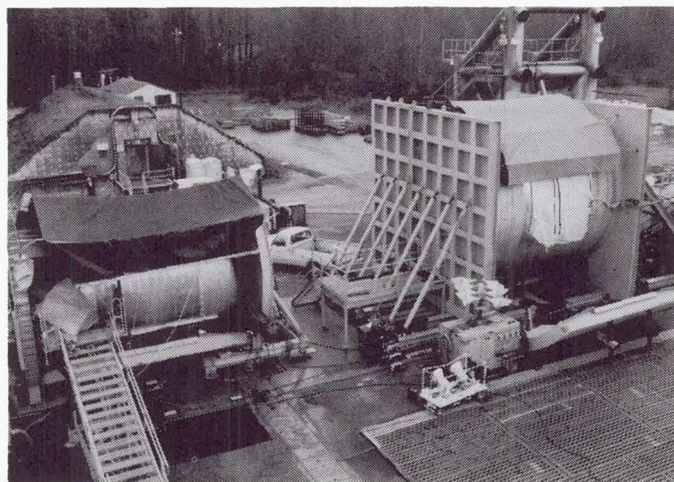


Figure 21. Boeing Pressure-Test Fixtures.

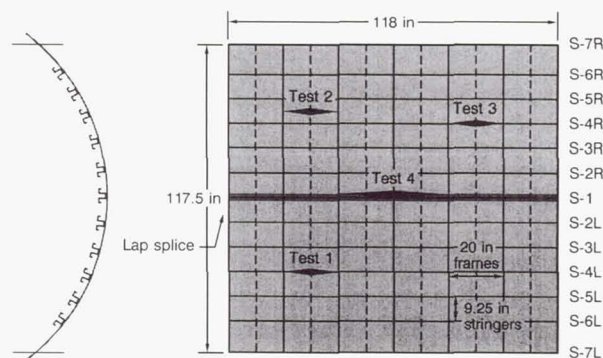


Figure 22. Typical Pressure-Test Panel.



Although attention had been given to a structural fuse system, damage was sustained during the first test when the test panel reached dynamic failure. Since then, the fixture has been repaired and the fuse arrangement modified.

To date, the new system has protected the fixtures from further damage during six dynamic panel failures, three on each test fixture. Damage tolerance testing has been done on 737 panels, 777 developmental panels, and on generic panels (Figures 23, 24, 25 respectively).

The 737 tests on the large pressure-test fixtures represented panel production configurations and field repair of a disbanded waffle doubler. Four of the tests were in lap joints. From these data, three major observations were derived:

- All panels were able to sustain at least a one-bay crack at operating pressure, including those with small saw cuts in the lap joints, simulating conservative MSD.
- Damage at the lap joint location, which produced safe decompression by flapping without MSD, caused dynamic failure after a one-bay crack when a generous degree of MSD was added.
- All tests conducted outside the lap splice provided safe decompression by flapping, except one panel that was overpressurized by 17%. That panel failed dynamically.

The 777 test panel configurations were experimental and may not represent the design actually selected for production. All panel configurations tested were fully shear-tied. The primary differences between panels were in the materials used; small changes in thickness of skin, shear ties, and frames; and a few other details.

The data showed that all panels were able to sustain operating pressure with two-bay cracks or greater, with the central frame severed (Figure 24). The latest panel tested as of this writing held a 20% overpressure without failure. One panel was tested with a saw cut in the lap joint without MSD and performed in approximately the same manner as the non-lap splice locations.

Panel configuration - narrow body pressure test fixture	Maximum crack length sustained at operating pressure	Residual strength results
• Riveted repair of tear straps Centered mid bay	1 bay (~20 in)	Dynamic at 17% overpressure
• Riveted repair of tear straps Centered mid bay	1 bay (~20 in)	Flapped
• Riveted repair of tear straps Centered on frame	1 bay (~20 in)	Flapped
• Bonded tear straps Centered mid bay	1 bay (~20 in)	Flapped
• Bonded tear straps Centered on frame	1 1/2 bays (~30 in)	Flapped
• Blind riveted repair of tear straps Centered mid bay	1 bay (~20 in)	Flapped
• Blind riveted repair of tear straps Centered on frame	1 1/2 bays (~30 in)	Flapped
• Lap joint Riveted repair of tear straps	2 bays (~40 in)	Flapped
• Lap joint with simulated MSD Riveted repair of tear straps	1 bay (~20 in)	Dynamic
• Lap joint with broken central frame Riveted repair of tear straps	1 bay (~20 in)	Dynamic
• Lap joint with simulated MSD Bonded tear straps	1 bay (~20 in)	Dynamic

Figure 23. Damage Tolerance Testing—737.

Panel configuration - wide-body pressure test fixture	Maximum crack length sustained at operating pressure	Residual strength results
• Broken central frame	2 bays (~40 in)	Repaired
• Broken central frame	Over 2 bays (~58 in)	Dynamic
• Lap joint - broken central frame	Over 2 bays (~48 in)	Repaired
• Broken central frame	2 bays (~40 in)	Repaired
• Broken central frame	Over 2 bays (~55 in)	Dynamic
• Broken central frame	2 bays (~40 in)	Held 20% over pressure

Figure 24. Damage Tolerance Testing—777 Development.

Panel configuration - wide-body pressure test fixture	Maximum crack length sustained at operating pressure	Residual strength results
• Bonded tear straps - 0.071 in skin Crack near stringer	1 bay (~20 in)	Repaired
• Bonded tear straps - 0.063-in skin Crack between stringers	1 bay (~20 in)	Repaired
• Bonded tear straps - 0.063-in skin Crack near stringer	1 bay (~20 in)	Repaired
• Bonded tear straps - 0.063-in skin Lap splice with simulated MSD and broken frame	2 bays (~40 in)	Held 9.5 psi prior to cutting frame Dynamic at 9.0 psi with cut frame
• Shear ties and tear straps Crack near stringer	1 bay (~20 in)	Repaired
• Bonded tear straps Crack between stringers	1 bay (~20 in)	Repaired
• Bonded tear straps Crack near stringer	1 bay (~20 in)	Repaired
• Lap splice with simulated MSD and broken frame Shear ties and tear straps	2 bays (~40 in)	Dynamic at 9.4 psi

Figure 25. Damage Tolerance Testing—Generic.

The two generic pressure-test panels were each tested at four locations (Figure 25). The first three locations were tested to collect load redistribution data from strain gages and panel failure was not expected. The focus of tests on both of these panels was the residual strength of the lap splice with MSD.



Small saw cuts, with a distribution of depths up to 0.020 inch, were made on the center four frame bays prior to panel assembly. Each panel sustained at least a two-bay crack in the lap splice with the central frame severed at normal operating pressure. The panels were then overpressurized until failure.

## CONCLUSIONS

The following conclusions were made:

- Guillotine tests, started in the 1950s, have shown that pressure structure reinforced with tear straps and/or shear ties can safely sustain a large amount of damage at any location.
- At locations where fatigue cracks are likely to develop in a row of fastener holes, small cracks under 0.03 inch at all holes can significantly reduce the residual strength of a large crack in the same row of holes.
- All configurations tested, including those with MSD in adjacent holes, show the structure can safely contain at least a one-bay crack of approximately 20 inches. Configurations with at least 19% stiffening tear straps and/or full shear ties can contain a two-bay skin crack with a severed central frame. However, widespread fatigue damage in adjacent bays could reduce these critical size cracks.
- Although all testing has shown that typical fuselage pressure structure can sustain at least a 20-inch crack, the best opportunity for safely detecting a crack in a longitudinal skin splice is before the crack reaches 1 to 2 inches. Detecting cracks in their early stages takes best advantage of the long time period (5 to 20 years) between detection and linkup. Also, as the crack becomes longer, the likelihood of widespread fatigue cracking in adjacent structure increases, resulting in reduced residual strength.
- Cracks in skin gages of 0.040 or less, reinforced with tear straps and/or shear ties, show a strong tendency to form flaps and provide safe decompression, except when the cracks appear in a row of fasteners containing a large (and perhaps unrealistic) amount of MSD. Cracks in skin gages 0.63 inch and greater have not demonstrated this tendency to form flaps.

## SUMMARY

Thirty-five years of testing for performance of fuselage pressure structure have now been reviewed. Saw cut and guillotine tests have shown a substantial and adequate amount of damage can be contained when the skin of an aircraft is reinforced with tear straps and shear ties, as is standard at Boeing and typical of the industry today.

These data show further that current standards are satisfactory for areas of the fuselage pressure structure and are not susceptible to widespread MSD. However, the fuselage longitudinal splices are susceptible to widespread MSD and need special consideration.

The 737 panel testing has demonstrated that conservative amounts of MSD may prevent the formation of flaps and, therefore, prevent safe decompression. In addition, flat panel tests indicate that MSD may reduce the residual strength for heavier skin gages that do not tend to form flaps.

The 707 full-scale fatigue test and the 737 aft fuselage fatigue test have shown that safe decompression occurs by flapping and that naturally-generated MSD did not prevent flapping. The concern is that a worst-case MSD due to normal fatigue scatter could prevent flapping and could also reduce the residual strength for heavier skin gages that do not tend to flap. These concerns dictate that an inspection program is required to detect cracks before they start to interact with cracks in adjacent frame bays.

Examination of the crack growth data from the 737 aft body test and service, the 747-100SR test, and the 747-400 test indicates a damage detection period of 5 to 20 years between detection (using current practical technology) and linkup or near linkup. Further, an additional safe damage detection period exists after linkup. The overall time period available to safely detect cracks in longitudinal joints is adequate provided the necessary inspections are identified and performed.

## RECOMMENDATIONS

In light of the above tests, the following recommendations are made:

- Even in cases where an adequate damage detection period is available, structural modification should be considered as an alternative to an intensified inspection program after the structure has reached a service threshold for possible MSD initiation. This modification has the advantages of eliminating the risk of the inspections not being adequately performed and providing timely scheduling of a "fix," which is likely to become necessary.
- To address worst-case scenarios, service thresholds for possible MSD initiation and the subsequent damage detection period should be based on the disbanded condition of longitudinal fuselage splice areas. Disbands in lap splices have occurred in service resulting in significant loss of fatigue performance and faster crack growth rates.

# FRACTURE MECHANICS RESEARCH AT NASA RELATED TO THE AGING COMMERCIAL TRANSPORT FLEET

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## SUMMARY

NASA is conducting the Airframe Structural Integrity Program in support of the aging commercial transport fleet. This interdisciplinary program is being worked in cooperation with the U. S. airframe manufacturers, airline operators, and the FAA. Advanced analysis methods are under development and an extensive testing program is under way to study fatigue crack growth and fracture in complex built-up shell structures. Innovative nondestructive examination technologies are also being developed to provide large area inspection capability to detect corrosion, disbonds, and cracks. This paper reviews recent fracture mechanics results applicable to predicting the growth of cracks under monotonic and cyclic loading at rivets in fuselage lap-splice joints.

## INTRODUCTION

On April 28, 1988, an Aloha Airlines Boeing 737 experienced an inflight structural failure when the upper fuselage ripped open and a large section of the skin peeled away. This failure was precipitated by the link-up of small fatigue cracks extending from adjacent rivet holes in a fuselage lap-splice joint. This event, brought about by multi-site damage (MSD), helped focus the attention of the industry on the problems of operating an aging commercial transport fleet. Currently, about 46 percent of the jet airplanes in the fleet are over 15 years old, with 26 percent being over 20 years old. During the past two years the industry has acted to ensure the continued safe operation of the aging fleet. These activities included increased emphasis on maintenance, inspection, and repair as well as mandatory modifications to various models in the fleet. Additional ways of ensuring safety are being vigorously pursued for both the current fleet and aircraft for the next-generation fleet.

This paper describes the research activities of the NASA Airframe Structural Integrity Program (ASIP) which has the goal of developing improved technology to support the safe operation of the current fleet and the design of more damage-tolerant aircraft for the next-generation fleet. Basic research related to fatigue and fracture of metals, computational fracture mechanics, structural analysis methods, and nondestructive examination (NDE) methods for material defect characterization has been ongoing at NASA Langley for many years. All of these disciplines have been brought to bear on the problems facing the aging commercial transport fleet. NASA has developed the ASIP in coordination with the FAA and the U. S. airframe manufacturers. The ASIP has two key program elements. They are the development of advanced analysis methodology to predict fatigue crack growth and residual strength of complex built-up structure and innovative NDE technologies to detect cracks, corrosion, and disbonds in adhesively bonded joints. These key elements will be verified



with an extensive test program on simple laboratory specimens and on built-up structure similar to the lap-splice joint region. The near term focus of the program is MSD in lap-splice joints and crack growth under mixed-mode conditions. However, the research is generic in nature and the developed methodology is expected to be applicable to many other structural components that may be fracture critical.

The objective of the analysis methodology program is to develop and verify advanced mechanics-based prediction methodology which can be used to determine inservice inspection intervals, quantitatively evaluate inspection findings, and design and certify damage-tolerant structural repairs. This objective will be met by developing an analysis methodology that integrates global shell analysis with local fracture mechanics analysis to predict fatigue crack growth and fracture characteristics in a fuselage structure. This can best be accomplished by developing and exploiting global/local strategies for combining the necessary levels of modeling and analysis. These levels are shown schematically in Figure 1. A sufficiently detailed local analysis is used to obtain the boundary conditions for a physically meaningful yet computationally tractable crack problem. Then a fracture mechanics analysis employing local crack-tip mechanics is used to predict fatigue crack growth and residual strength. The effects of the crack growth must then be integrated upward to the global structural level to insure that correct load transfer paths and internal load distributions are calculated. Of course these analyses may have to be performed in an iterative fashion to achieve correct results.

The program logic is shown schematically in Figure 2. The individual program elements are discussed in more detail in the following sections. A brief overview of the direction of research for each element will be followed by a summary of the status of the work in progress. Relevant results will be discussed in areas where the research is sufficiently mature for presentation.

#### FATIGUE CRACK GROWTH AND CLOSURE MODEL

The concept of crack closure to explain crack growth acceleration and retardation was pioneered at NASA Langley almost two decades ago [1]. The closure concept, illustrated schematically in Figure 3, is based on the postulate that the wake of plastically deformed material behind an advancing crack front may prevent the crack from being fully open during the complete loading cycle. Therefore, only part of the load cycle is effective in growing the crack. A plasticity-induced closure model [2] employing fracture mechanics principles was shown to be quite accurate in predicting fatigue crack growth in aluminum alloys for a number of basic crack configurations for both constant amplitude and spectrum loadings. The closure model has also been successfully used to explain the extreme crack growth behavior during proof testing [3] and the small-crack growth phenomenon exhibited by many aluminum alloys [4]. The crack growth rate data must be correlated with the effective stress-intensity factor range rather than the full range to give meaningful predictions of crack growth. The successful coupling of the closure methodology with the small-crack growth rate data base has resulted in a total life prediction methodology which treats initiation by predicting the growth of micron-size cracks initiating at inclusion particles in the subgrain boundary microstructure [5]. This type of methodology is necessary to predict the growth of small cracks initiating at rivet holes before they grow to a detectable size. Furthermore, this methodology may be used to predict the necessary inspection intervals to monitor crack growth before critical sizes are reached and link-up of adjacent cracks occur.



## Recent Results of Work-in-Progress

A computer code, FASTRAN, has been developed for predicting crack growth using an analytical closure model and the code is available in the public domain through COSMIC [6]. The present version FASTRAN-II, with improved spectrum load input options, was developed to run on mainframe and personal computers [7]. An example of the accuracy and computational efficiency of this code was demonstrated recently by comparing predicted lives with experimental lives from tests on center-crack tension coupons made of 7075-T6 aluminum alloy sheet subjected to the Mini-TWIST spectrum. These tests were conducted by Mr. E. P. Phillips at the NASA Langley Research Center. The Mini-TWIST spectrum is a standard transport wing spectra with about 64,000 loading amplitudes for 4,000 flights [8]. At a low mean stress,  $S_m = 20$  MPa, the experimental life (average of two tests) was determined to be 202,000 flights from an initial 6 mm length crack to failure. The crack length is plotted against the opening stresses calculated from FASTRAN-II in Figure 4. The predicted life was 20 percent higher than the experimental life. At a high mean stress,  $S_m = 60$  MPa, the experimental life (average of two tests) was determined to be 9,500 flights. In this case, the predicted life was 2 percent below the experimental life. The low and high mean stress cases required 16 and 1.3 CPU minutes, respectively, on a mainframe (CONVEX) computer using scalar and vector optimization. These predictions were based on calculating the opening stress during very small increments of crack extension throughout the life. However, by using a damage-weighted average opening stress,  $\bar{S}_0$ , the required computational time was reduced to 8 minutes and 1 minute for the low and high mean stress cases, respectively. A comparison between the model calculations and the average crack opening stress is shown in Figure 4. The predicted lives using  $\bar{S}_0$  were 12 percent higher and 3 percent lower than the experimental values for the low and high mean stress cases, respectively. The first application of the Mini-TWIST spectrum was used to create the characteristic plastic wake for the spectrum. Then the damage-weighted average opening stress was calculated from the closure model during the second time through the complete spectrum. The average value was then used for the repeated spectra until failure was predicted. This approach makes the closure model computationally cost competitive with empirical crack growth models presently used in the aerospace industry. The use of the PC version of FASTRAN-II will result in even further computational cost reductions but the computer times will be much larger than those from the mainframe computer. However, the rapid developments that are being made in computer science will shortly make the PC a vector and parallel processing machine with speeds competitive with the mainframe computer of today.

## THREE-DIMENSIONAL STRESS ANALYSES OF COUNTERSUNK HOLES

A riveted joint introduces discontinuities in the form of holes, changes in the load path due to lapping, and additional loads like rivet bearing and bending moments, as shown in Figure 5. Because of these complexities, local stresses are elevated in the joint. Accurate estimations of these local stresses are needed to predict fatigue life and joint strength.

Exhaustive studies on stress-concentration factors for holes and notches in two-dimensional bodies under a wide variety of loadings have been documented in handbooks. However, stress concentrations for holes in finite-thickness plates

have only been reported for a few configurations and loadings (circular and elliptical holes under remote tension and bending loads). Surprisingly, stress analyses of countersunk holes have not been reported and only a few experimental results are in the literature.

A comprehensive analysis of three-dimensional stress concentrations and stress-intensity factors for surface or corner cracks in circular straight-shank and countersunk holes under typical loads encountered in structural joints is being undertaken in the NASA ASIP.

#### Recent Results of Work-in-Progress

A three-dimensional finite-element stress analyses code (FRAC3D), developed at the NASA Langley Research Center using 20-node isoparametric elements, was used to obtain stress concentrations for a wide range in hole sizes (hole-radius-to-plate-thickness ratios) and countersink depths (from knife-edge to straight-shank). The straight-shank and countersunk hole configurations and loadings are shown in Figure 6. Three types of loads, namely: remote tension, remote bending and wedge loading were considered for the straight-shank hole. Two types of loads, remote tension and remote bending, were considered for the countersunk holes. The wedge-load solution is used in combination with the remote tension solution to obtain a simulated pin-load solution. Using the finite-element results, series-type equations were developed to give the stress concentration at any location along the bore of the holes. These results are reported in a paper by Shivakumar and Newman.\*

Some typical results for a countersunk hole with a straight-shank portion of 25 percent of the sheet thickness ( $b/t = 0.25$ ) for a wide range in hole-radii-to-thickness ( $r/t$ ) ratios are shown in Figure 7. The standard countersunk angle ( $\theta_c$ ) of 100 degrees was used in all cases. The stress concentration,  $K_t$ , is the ratio of local normal stress along the bore of the hole to the remote applied stress,  $S$ . As expected, the peak stress concentration occurred at the countersunk edge at a  $z/t$  value of  $-0.25$  (the  $z$  coordinate is measured from the center of the sheet). Results from the finite-element analyses are shown as symbols. The curves represent the equations that were developed to fit these results.\* The equations are useful in determining peak stress concentration factors when complex loading conditions, such as those encountered in service, are superimposed.

#### FRACTURE MECHANICS OF MULTI-SITE DAMAGE (MSD)

The MSD problem that has occurred recently in some high-time aircraft is generally associated with many neighboring cracks along a row of rivets, such as in the fuselage. However, MSD has not been confined to only the fuselage. Some early MSD problems developed in the wing structure. If a crack initiates at a single rivet hole, the displacement compatibility in the riveted joint will tend to unload the cracked hole and shed some load to its neighboring rivets. These adjacent holes will be overloaded and in turn develop cracks. All cracks tend to be of the same length except in regions near stiffeners or in regions of maximum pressure pillowing. This cracking and load redistribution process promotes the development of MSD. Thus, riveted lap-splice joints are prone to MSD if cracks ever develop. These joints should, therefore, be designed to prevent cracking or MSD from developing. However, to develop procedures to design against MSD, methods should be developed to predict initiation of cracks

\*Shivakumar, K. N. and Newman, J. C., Jr.: Stress Concentrations for Straight-Shank and Countersunk Holes in Plates Subjected to Tension, Bending and Pin Loads, Submitted for publication as a NASA TP, National Aeronautics and Space Administration, Washington, D. C.



and to analyze the complex problem of interacting cracks. Of course, a key element is an accurate stress analysis of the local stress conditions.

A rigorous fracture mechanics treatment of cracks initiating at rivet holes and MSD will require stress-intensity factor solutions for three fundamentally different levels of crack sizes. For very small cracks below the damage tolerance (inspectable size) regime, the finite-element method will be used to generate solutions to three-dimensional (3-D) crack configurations such as surface and corner cracks initiating at countersunk rivet holes, as shown in Figure 5. The anticipated results from these analyses will be a compendium of analytical expressions for the stress-intensity factors for several basic crack configurations. After the cracks extend through the wall thickness and beyond the rivet head, the cracks will be in the detectable range and amenable to the damage tolerance philosophy. A two-dimensional (2-D) indirect boundary element method is being used to generate stress-intensity factors for MSD crack configurations prior to extensive link-up, as shown in Figure 8. A PC-based computer code will be developed which will compute the stress-intensity factor for each crack tip for an unequal distribution of straight or curved cracks at adjacent rivet holes. After MSD crack link-up, the cracks are quite large and crack growth will be rapid. The stress-intensity factor for these cracks will be strongly influenced by the geometric nonlinear response of the stiffened fuselage shell structure. A geometric nonlinear finite-element shell code will be developed that will model the stiffening effects of longitudinal stiffeners and circumferential frames. The crack-tip stress-intensity factors will be calculated by employing special crack-tip modeling and an adaptive mesh strategy to properly model the trajectory of the growing crack. Research is underway at NASA to develop methods to rigorously and efficiently address all three levels of crack growth.

During the last decade, NASA has developed several computational methods for computing stress-intensity factor solutions to complex crack configurations. The Boundary-Force Method (BFM) [9] is well suited to analyze 2-D problems such as through cracks in thin sheet material. An example of the type of problem that can be efficiently treated by the BFM method is illustrated in Figure 9. This specimen, and a three-hole cracked specimen [10], was designed to be a simple specimen to simulate the results of a cracked stiffened panel. The stress-intensity factor solution is remarkable close to that for a stiffener located along the hole centerline [11].

For more complex problems, the finite-element method with the nodal-force method [12] and the virtual-crack-closure technique (VCCT) [13] have been successfully employed to obtain solutions to many 3-D crack configurations. One example, for a surface crack emanating from a semi-circular notch, is shown in Figure 10. Both methods give essentially the same results. A discrepancy occurs where the crack front intersects the notch (or free) boundary at a parametric angle  $\varphi$  of 90 degrees because of a boundary-layer effect. Empirical stress-intensity factor equations, such as that shown by the solid curve, have been developed for many of the 3-D crack configurations [14].

The equivalent domain integral method (EDI) [15,16] has also recently been implemented into several NASA finite-element codes. This technique is well suited for 2-D and 3-D crack problems involving mixed-mode loadings and material nonlinearity. For example, the J-integral computed from the EDI method is compared to analytical results from the VCCT [13] for an elastic problem in Figure 11. Both techniques gave nearly the same values of the J-integral and they compare well with the equation developed by Newman and Raju [14]. An additional advantage of the EDI method is that it does not require that the



finite-element mesh be orthogonal to the crack front, as required in the nodal-force method, to calculate stress-intensity factors. Both the VCCT and EDI methods have been implemented into a public domain computer code, ZIP3D [17], available through COSMIC.

NASA has developed empirical stress-intensity factor equations for many basic 3-D crack configurations by fitting to the various finite-element results. Reference 14 gives equations for many of the crack configurations commonly encountered in structural applications.

### Recent Results of Work-in-Progress

The near term focus of the NASA fracture mechanics research is on MSD prior to link-up to support the damage tolerance philosophy currently being implemented by the airline industry. An Alternating Indirect Boundary Element (AIBE) computer code [18], a force method version of the more familiar boundary-element method, is being developed for MSD crack configurations. The AIBE code will generate stress-intensity factors for individual cracks in an arbitrary distribution of unequal, interacting cracks extending from loaded rivet holes. Methods are being developed so that in-plane, out-of-plane, and rivet loads can also be included in the analysis. The current version of the code can only model straight cracks. However, a 2-D approach to analyze curved cracks is being developed using the distributed dislocation concept [19,20]. This method will also be available to analyze MSD crack configurations. Additionally, a special version of FASTRAN has been coupled to the AIBE code so that fatigue crack growth can also be computed.

Each new capability of the AIBE code is being verified by comparison to available analytical results from other computational techniques for simple crack configurations. A complete code verification will be achieved by comparing analytical predictions to experimental results. To date, AIBE code predictions have been compared to experimental results for the special case of multiple fatigue cracks growing from open holes in thin sheets subjected to constant-amplitude remote uniform tension loadings. Figure 12 shows a typical experimentally measured distribution of crack lengths along with the stress-intensity factors calculated for each crack. The stress-intensity factors are normalized by the stress-intensity factor for an average crack length assuming an infinite periodic array of equal cracks. Figure 13 shows comparisons between fatigue crack growth predictions for the unequal distribution of cracks, the predictions for an infinite periodic array of equal cracks, and experimental test results. It is obvious from these results that the simplistic approach of assuming equal crack lengths may not always generate accurate predictions of crack growth and cycles to crack link-up. Many additional experimental cases of crack configurations and loading conditions are being conducted in order to refine the AIBE code and to more fully verify its accuracy.

### NONLINEAR STIFFENED-SHELL STRUCTURAL ANALYSIS

The behavior of large cracks in fuselage structures such as midbay cracks or splice joint cracks after MSD link-up are strongly influenced by the stiffening effects of the circumferential frames and longitudinal stiffeners. The consequences of multiple-element damage (MED), such as stiffener cracking, on the behavior of large cracks in the fuselage should also be accounted for in the analysis. It is not practical to model all of the structural details in a finite-element analysis. Greater efficiency can be achieved by exploiting a global/local strategy where local details that produce stress gradients can be

treated in a companion analysis to the global structural analysis. The structural analysis methodology will have to account for geometric nonlinear behavior as well as large deformation behavior. Effects such as pressure pillowing, the outward bulge of the skin between stiffeners, must also be accounted for in the analysis. This is necessary to predict the crack- growth direction and crack opening of large cracks that may result in a rapid decompression rather than a catastrophic in-flight failure.

### Fracture Analysis with Adaptive Meshing

To predict accurately the behavior of a growing crack in a stiffened shell structure, the global/local methodology must be extended to include an adaptive mesh concept so that the local refined mesh can change in a manner dictated by the growing crack. This concept is illustrated schematically in Figure 14 taken from the work of Wawrzynek and Ingraffea [21]. The path of the growing crack is represented by the heavy dark line in Figure 14. The top two schematics illustrate the element deletion and refill concept while the lower two schematics illustrate a crack growth example problem.

### Methodology Verification Test Program

The integrated fracture mechanics and fuselage structural analysis methodology must be verified by a test program. As shown in Figure 15, there are various levels of testing required to achieve a full verification of a structural analysis methodology. The goal of the ASIP program is to achieve verification through the curved panel and subscale barrel test article level. This program element will be conducted in cooperation with Boeing Commercial Airplane Company and Douglas Aircraft Company. Each airframe manufacturer will contribute flat panels with riveted splice joints typical of their manufacturing practices. Furthermore, data from previous fatigue and damage tolerance test programs conducted by the airframers will serve as benchmarks for the analysis methodology verification.

### CONCLUDING REMARKS

NASA is conducting interdisciplinary research combining the disciplines of fracture mechanics, structural mechanics, material science, and nondestructive instrumentation sciences for the purpose of developing an advanced integrated technology to support the continuing airworthiness of the aging commercial transport fleet. The objective of the analysis methodology and test program is to develop and verify advanced mechanics-based prediction methodology which can be used to determine inservice inspection intervals, quantitatively evaluate inspection findings, and design and certify damage tolerant structural repairs. The program is coordinated with the FAA and is being worked cooperatively with the U. S. airframe manufacturers and airline operators. Fracture mechanics solutions to cracks extending from the rivets in fuselage splice joints are being developed which will be applicable to multi-site damage (MSD). A plasticity-induced closure model is being developed which will be applicable to predicting the fatigue crack growth of MSD crack configurations. This fracture mechanics methodology is being integrated into a finite-element based stiffened-shell structural analysis methodology so that significant geometric nonlinear effects and MED on crack growth may be accurately predicted. The integrated methodology will be verified by analyzing fatigue and damage tolerance test results on curved panels with stiffeners and frames.



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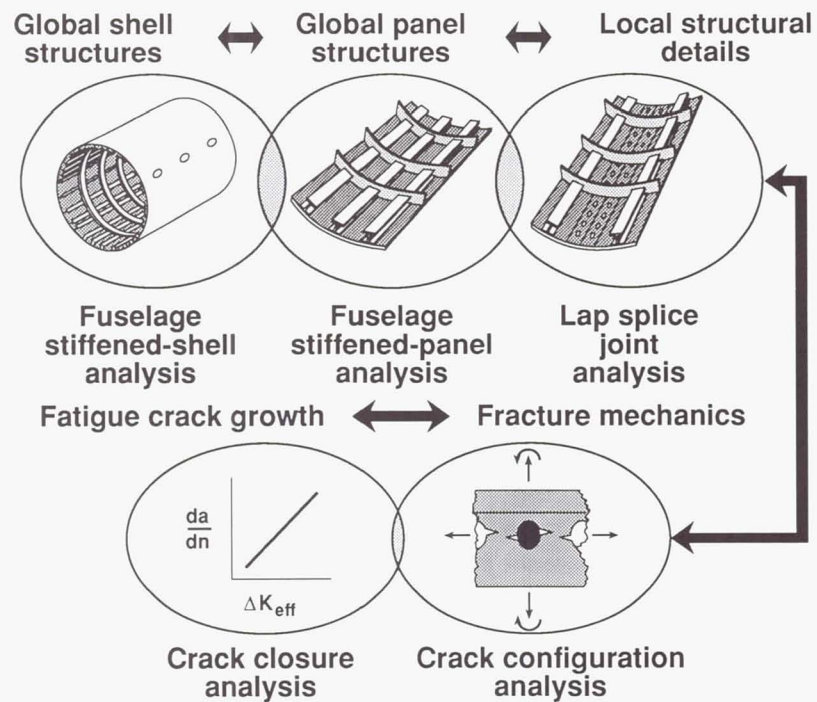


Figure 1. Integrated shell analysis-fracture analysis methodology.

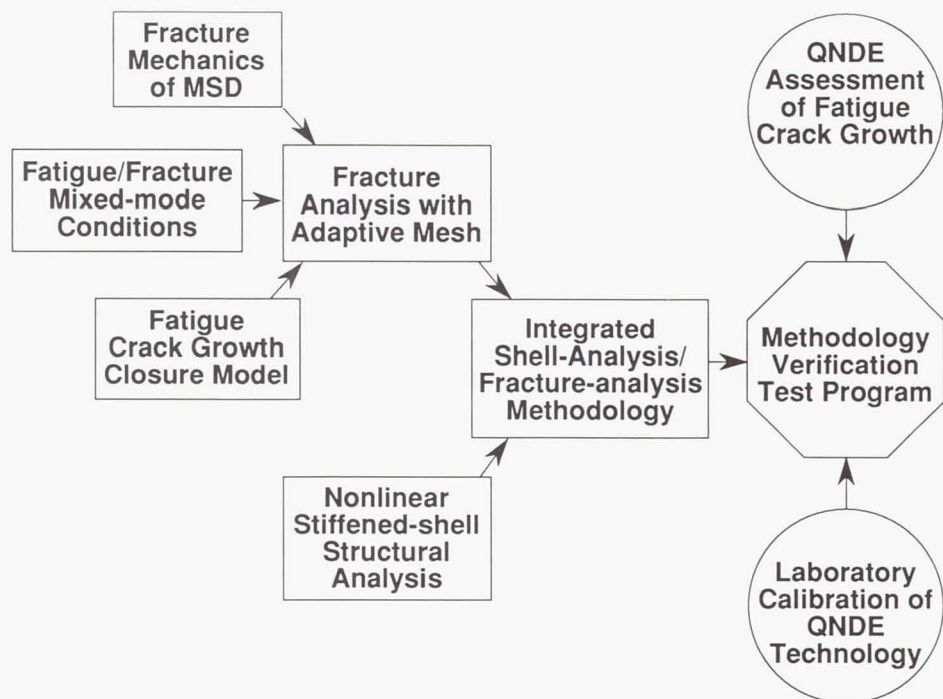


Figure 2. Logic diagram for analysis methodology program.

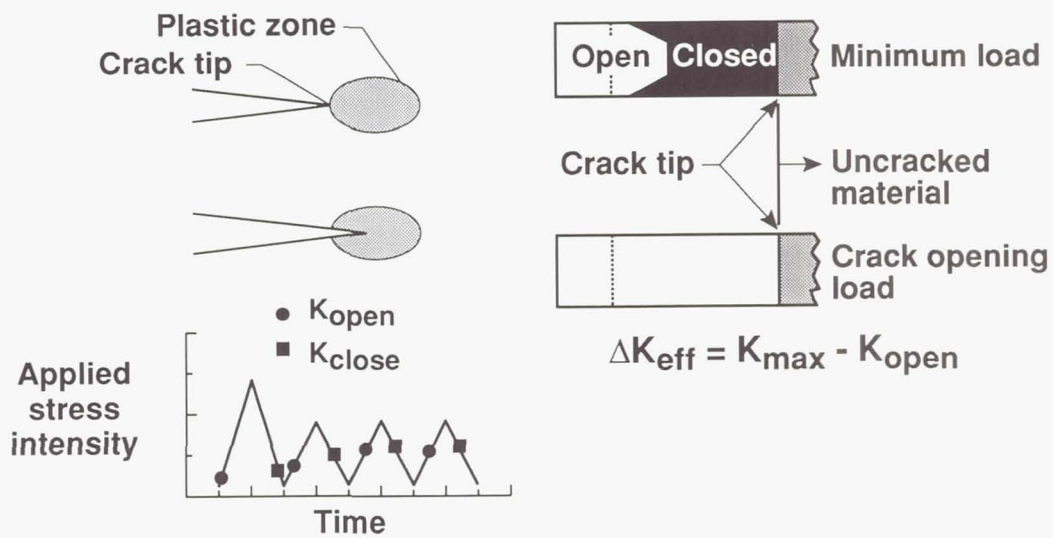


Figure 3. Fatigue crack growth controlled by closure mechanism.

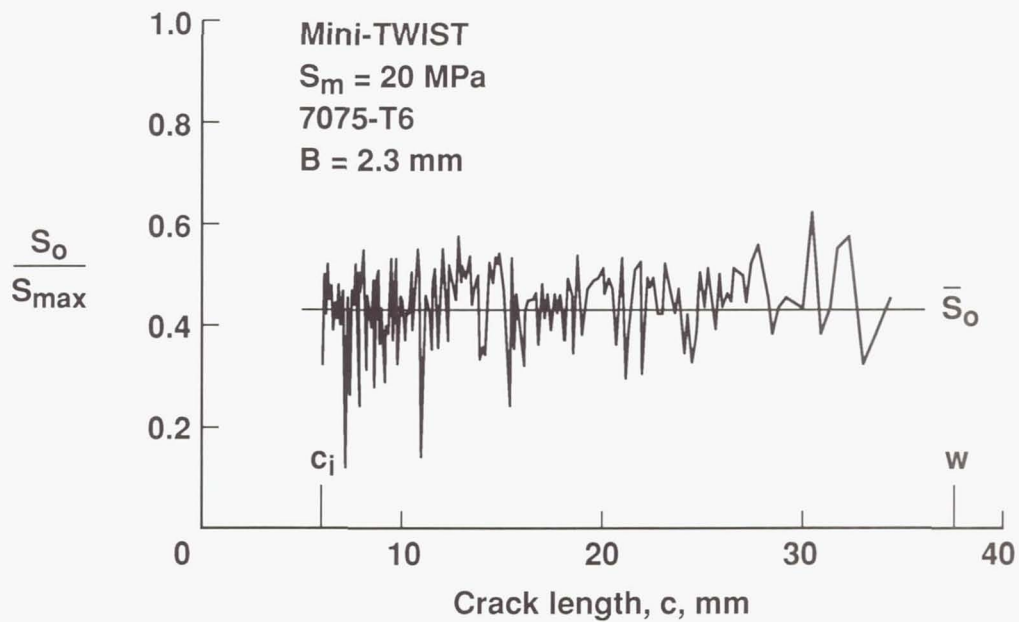


Figure 4. Crack-opening stress variation during flight-by-flight loading.



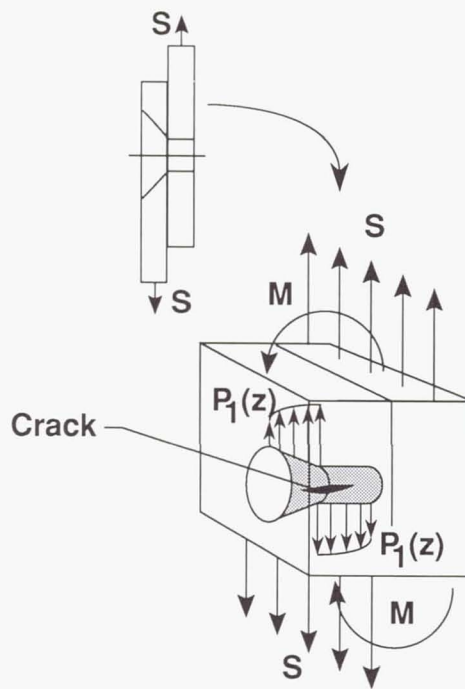


Figure 5. Cracking and loading conditions encountered at countersunk rivets.

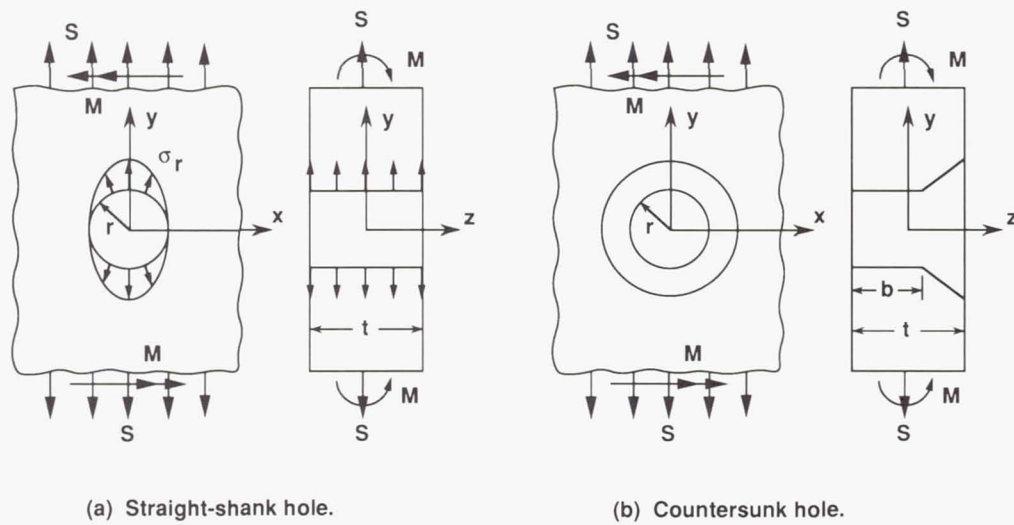


Figure 6.- Straight-shank and countersunk hole configurations and loadings.

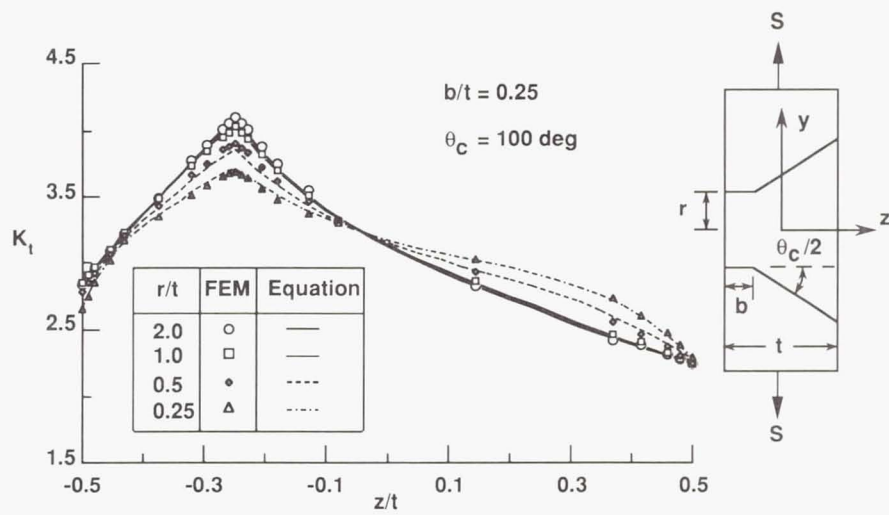


Figure 7. Comparison of stress concentration factors from finite-element analyses and equations for countersunk rivet hole under remote tension.

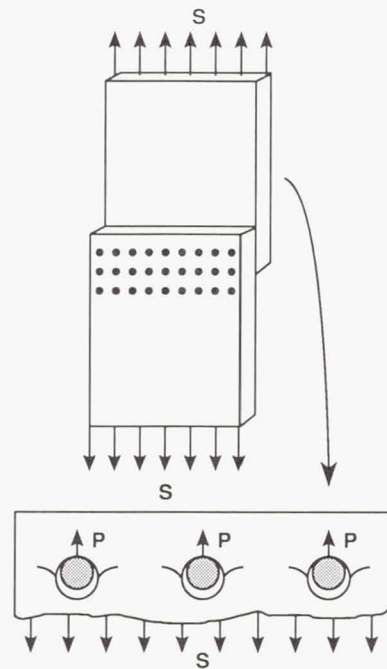


Figure 8. Fracture mechanics analyses of multiple-site cracks.

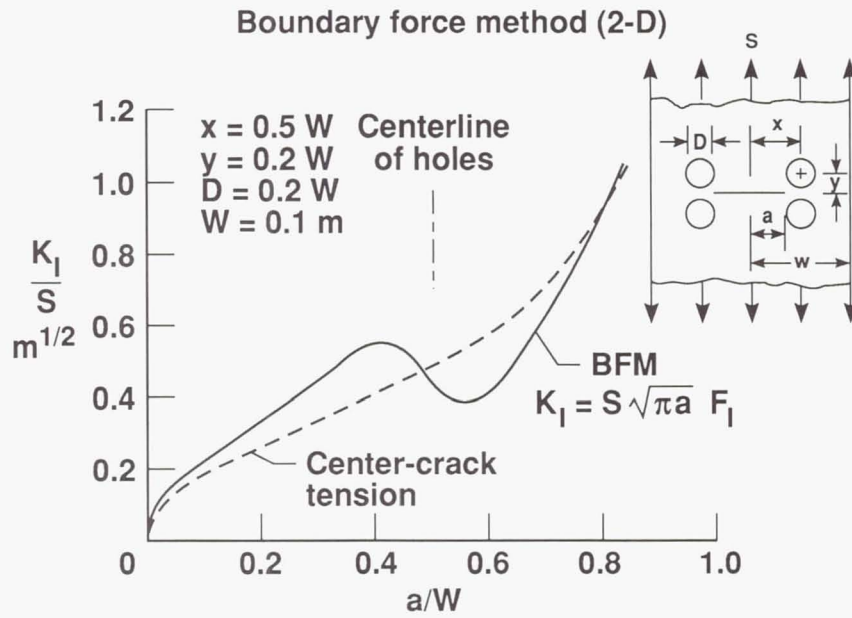


Figure 9. Stress-intensity factors for a four-hole cracked specimen.

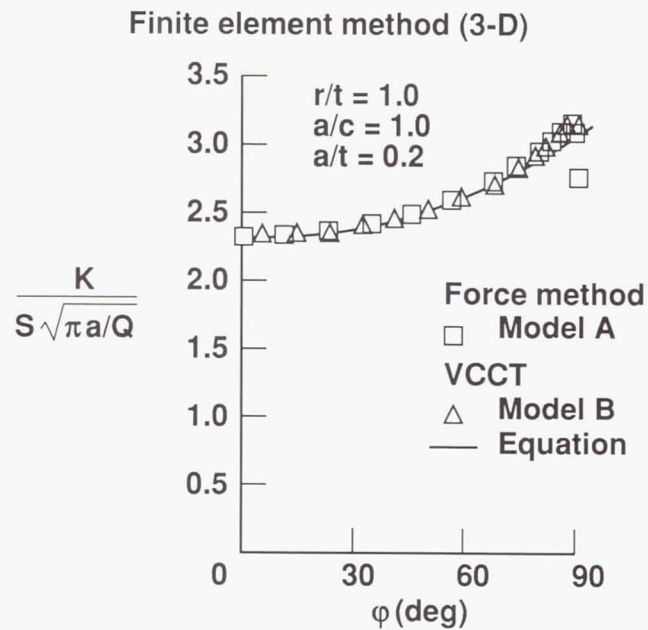


Figure 10. Stress-intensity factors for a surface crack emanating from a semi-circular notch.



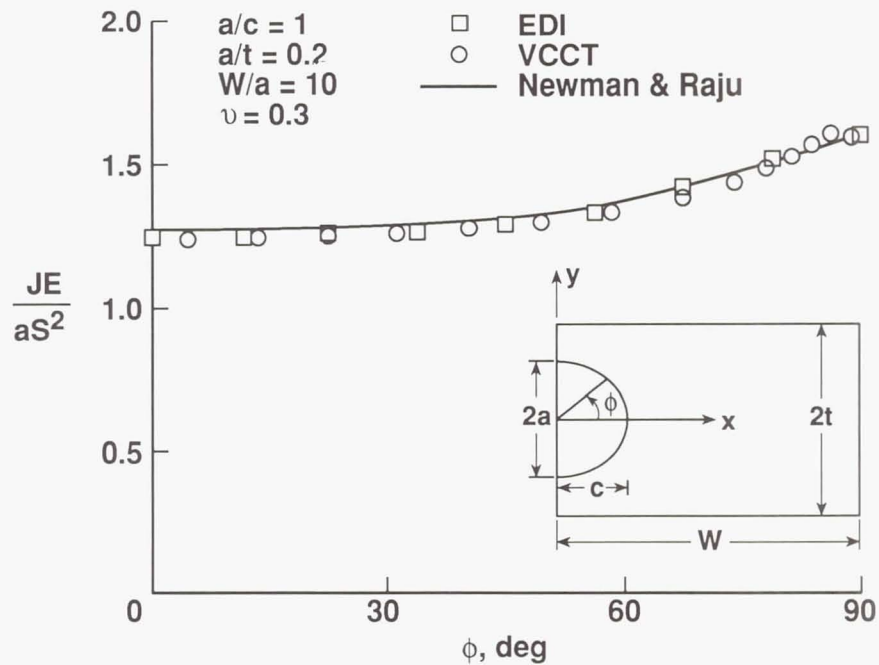


Figure 11. Comparison of normalized J distribution for a semicircular surface crack from EDI, VCCT, and force methods.

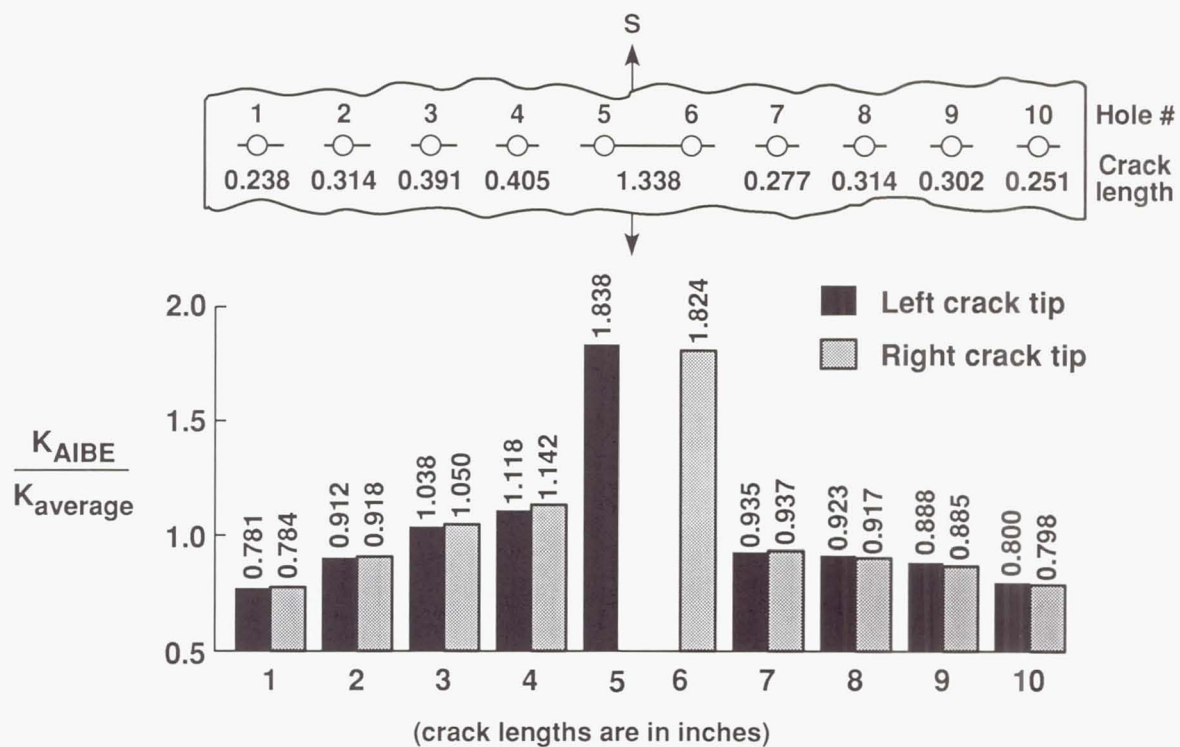


Figure 12. AIBE determined stress intensity factor distribution for an open hole MSD experiment.

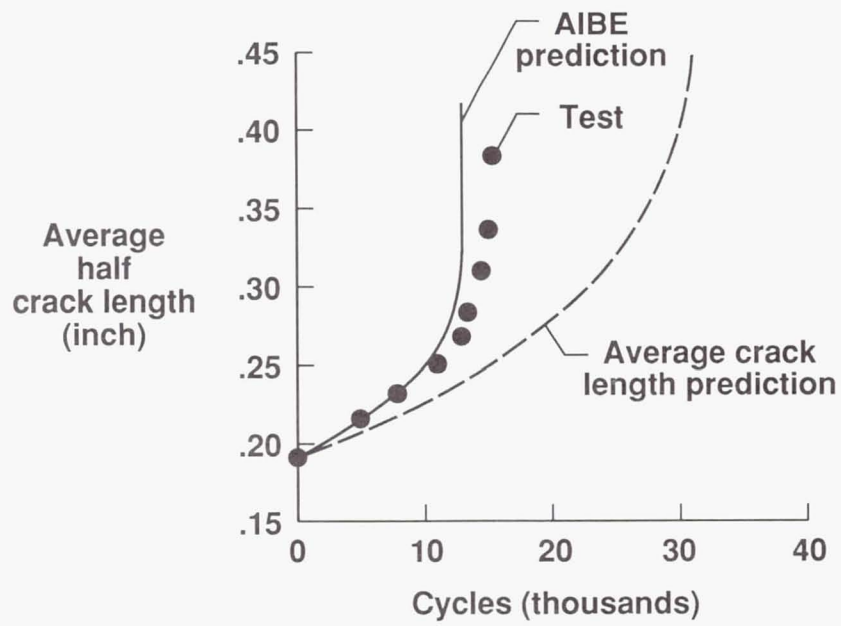


Figure 13. Predicted fatigue crack growth behavior for an open hole MSD experiment.

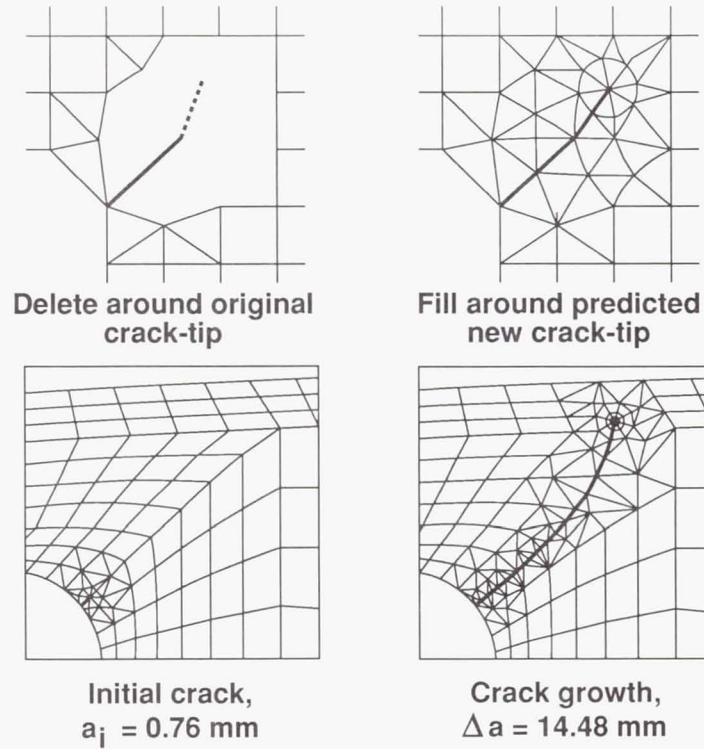


Figure 14. Crack propagation by adaptive remeshing using the finite element method.

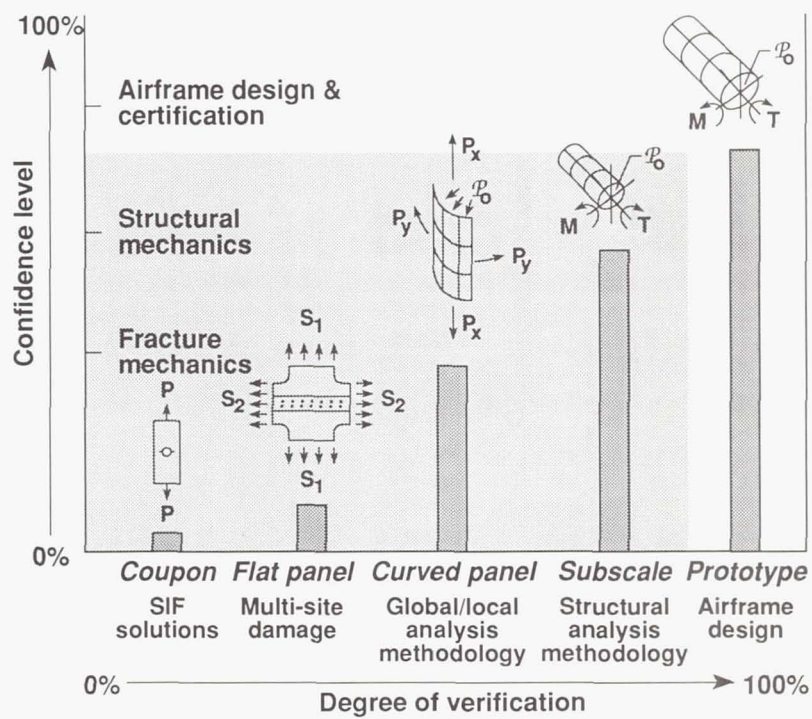


Figure 15. Prediction methodology verification program.



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# PRELIMINARY RESULTS ON THE FRACTURE ANALYSIS OF MULTI-SITE CRACKING OF LAP JOINTS IN AIRCRAFT SKINS

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## INTRODUCTION

This paper presents results of a fracture mechanics analysis relevant to fatigue crack growth at rivets in lap joints of aircraft skins. Multi-site damage (MSD) is receiving increased attention within the context of problems of aging aircraft. Fracture analyses carried out previously include small-scale modeling of rivet/skin interactions, larger-scale two-dimensional models of lap joints similar to that developed here, and full scale three-dimensional models of large portions of the aircraft fuselage. Fatigue testing efforts have included flat coupon specimens, two-dimensional lap joint tests, and full scale tests on specimens designed to closely duplicate aircraft sections. Most of this work is documented in the proceedings of previous symposia on the aging aircraft problem. Other sources include reference 1, which provides a detailed summary of final results from the testing and finite element analysis of full-scale models of aircraft sections. References 2 and 3 offer analyses of the interaction of large-scale aircraft skin cracks with stiffeners in the absence of MSD. The effect MSD has on the ability of skin stiffeners to arrest the growth of long skin cracks is a particularly important topic that remains to be addressed.

One of the most striking features of MSD observed in joints of some test sections and in the joints of some of the older aircraft fuselages is the relative uniformity of the fatigue cracks from rivet to rivet along an extended row of rivets. This regularity suggests that nucleation of the cracks must not be overly difficult. Moreover, it indicates that there is some mechanism which keeps longer cracks from running away from shorter ones, or, equivalently, a mechanism for shorter cracks to "catch-up" with longer cracks. This basic mechanism has not been identified, and one of the objectives of the present work is to see to what extent the mechanism is revealed by a fracture analysis of the MSD cracks. Another related aim is to present accurate stress intensity factor variations with crack length which can be used to estimate fatigue crack growth lifetimes once cracks have been initiated. Results will be presented which illustrate the influence of load shedding from rivets with long cracks to neighboring rivets with shorter cracks. Results will also be included for the effect of residual stress due to the riveting process itself. All the results presented in this paper should be regarded as preliminary in the sense that further work by ourselves and

others will be required to validate them and to establish their sensitivity to more embellished modelling assumptions.

## MODEL

The model is defined with reference to figures 1 and 2. The joint is imagined to be infinite in width with three rows of rivets equally spaced a distance  $2w$  apart in the  $x$ -direction. The vertical spacing between the rows is  $h$ , and the rivet radius is  $R$ . The average stress carried across the joint is  $\Sigma_{yy}$ . As indicated in figure 1, a straight crack of length  $a$  emerges from the horizontal extremity of each side of each rivet hole in the top row of the outside skin. Each section of width  $2w$  containing a line of 3 rivets deforms identically, and thus attention can be directed to the behavior of just one section, such as that in figure 1. Furthermore, each section is symmetric with respect to the plane  $x=0$ , so that only one half of a section needs to be modelled, as shown in figure 2. Periodic boundary conditions must be enforced on the right vertical edges of the model. These require that the shear stress vanishes along the edges, and that the edges remain straight and parallel to the  $y$ -axis. The average strain in the  $x$ -direction (the relative horizontal displacement of the edges divided by  $2w$ ) must be such that the stress in the skin well away from the joint is  $\Sigma_{xx}$ . In this preliminary study, finite element calculations were made with  $\Sigma_{xx}=0$ ; however, the effects of nonzero transverse loading are approximated using a solution in the literature.

Bending out of the plane will be neglected. Each half of the lapped skin is modeled by a thin plate, or sheet, in plane stress with load transferred from one to the other through rivets which are modeled in a manner discussed below. Three-dimensional aspects of the rivet geometry, such as countersinking, are ignored. Residual stress in the rivet and skin in the vicinity of the rivet hole due to the riveting process is likely to effect fatigue crack growth, and an initial attempt to assess this effect is included in a subsequent section of the paper. Bonding by an adhesive layer between the lapped sheets is not considered. Plasticity is not taken into account; deformations are assumed to be elastic. Some of the effects being ignored are likely to be important, and it is our intention to include them in a more elaborate model in the near future.

The next section presents stress concentration factors for the uncracked joint after several ways to model the rivet/skin interaction have been discussed. The sections following then present stress intensity factors for the MSD cracks, an approximate analysis of the role of residual stress induced by the riveting process, and preliminary work related to load shedding from rivets with longer cracks to rivets with shorter cracks. The calculations reported below have been carried out using dimensions of a typical aircraft lap joint with  $w=0.50$  in.,  $h=1.00$  in.,  $R=0.0805$  in., and  $d=0.50$  in. Thus, the nondimensional ratios used in the calculations are



$$h / R = 12.42, \quad w / R = 6.21, \quad d / R = 6.21 \quad (1)$$

The material properties used for aircraft skin were  $E=10.0$  Msi and  $\nu=1/3$ .

### STRESS CONCENTRATION AT THE RIVET HOLES IN THE ABSENCE OF CRACKS

A finite element model of the section shown in figure 2 has been developed using the ABAQUS code with eight-noded quadrilateral plane stress elements. Details of the rivet/sheet interaction will be shown to be very important. In the present study two simplified models of rivet/sheet interaction are used, in each case with rivets taken to be rigid with zero friction where they contact the boundary of the rivet hole. Thus, for both models, the displacements of the rivet center points A, B, and C for the outside skin (figure 2) were set equal, respectively, to those of rivet center points A', B', and C' for the inside skin. Some of the additional details of this modelling are described below. The effect of more elaborate modelling assumptions will be explored in subsequent work.

The problem of a rigid rivet contacting a rivet hole is, in general, nonlinear requiring iteration of some kind to determine the contact region. In this paper two models of the interaction will be pursued which we believe are limiting cases of the full range of behavior. One of the modelling assumptions is appropriate when there is little or no residual stress exerted by the rivet on the unloaded sheet so that the rivet separates from the hole when the joint is loaded (BC#1), and the other applies when the residual stress due to the riveting process results in the maintenance of contact between rivet and sheet around the entire circumference of the hole (BC#2). In the first case, in which the rivet is assumed to just fit into its hole with no residual stress acting across the hole boundary, exploratory calculations showed that under application of  $\Sigma_{yy}$  only the bottom half of the rivets contacted the holes of the outside sheet (and the top half of the rivets for the inside sheet), with the exception in a few cases where a very small contact zone existed just above the crack line in the outside sheet. Thus, for all the results presented in this section and the next section for the case BC#1, the contact region was assumed a priori to coincide with the lower half of the rivets for the outside sheet and the upper half of the rivets for the inside sheet. This boundary condition, together with the condition that the shear traction vanish over the contacting regions, is conveniently incorporated in the ABAQUS code. For the case in which residual stress forces contact over the entire circumference of the hole, the boundary condition is modified accordingly, retaining the assumption that shear tractions vanish everywhere around the hole. The effect of the residual stress distribution itself on the stress intensity factor of the crack will be taken up in the next section.

The mesh used when no cracks are present ( $a=0$ ) is shown in figure 3a. Care was taken to ensure that the computed stress concentration factors (SCFs) did not change with further refinement of the mesh. In the absence of the cracks, the behavior of the top and bottom sheets is identical and this was exploited to mesh just the outside sheet. In this single sheet model, the vertical displacement of point B is set equal to zero and the displacements of points A and C are constrained to be equal in magnitude and opposite in sign. The calculations for the SCFs, and for the stress intensity factors in the next section, were made with zero average transverse stress, i.e.  $\Sigma_{xx}=0$ . In a pressurized fuselage, a reasonable approximation for  $\Sigma_{xx}$  would be  $\Sigma_{yy}/2$ , but many tests are carried out with  $\Sigma_{xx}=0$ . In the next section, a simple approximation for including the influence of the transverse stress will be suggested.

With  $\sigma_{yy}$  as the stress component at the edge of the hole, directly to the left or right of its center, let  $SCF = \sigma_{yy}/\Sigma_{yy}$ . The SCFs at the holes in the outside sheet calculated using the rigid rivet model for the case BC#1 are

$$SCF_{\text{top hole}} = 4.43, \quad SCF_{\text{middle hole}} = 2.79, \quad SCF_{\text{bottom hole}} = 2.72 \quad (2)$$

The larger value at the top hole relative to the lower holes in the top sheet mainly reflects the fact the average of the stress component  $\sigma_{yy}$  carried by the top sheet is significantly higher above the top row of rivets than below it due to load transfer to the inside sheet. To examine the possible effect of rivet stiffness in shear, another calculation which takes the shear load carried by each of the three rivets to be the same (as would be expected in the extreme limit of very compliant rivets) has been carried out. It gives the following rather similar results

$$SCF_{\text{top hole}} = 4.25, \quad SCF_{\text{middle hole}} = 3.28, \quad SCF_{\text{bottom hole}} = 2.39 \quad (3)$$

Three-dimensional effects (e.g. countersinking and elasticity of the rivets) will effect local stress concentration factors in the vicinity of the holes before any cracks develop. The absolute values of the SCFs presented above are therefore not expected to be of particular significance. However, the relative levels do clearly indicate that fatigue cracks would be expected to start at the top hole of the outside sheet or at the bottom hole of the inside sheet, assuming that some additional factor such as countersinking did not differentiate between the two sheets.

#### ENERGY RELEASE RATE, STRESS INTENSITY FACTORS, AND COMPLIANCE CHANGE DUE TO CRACKING

As depicted in figure 2, cracks of length  $a$  are assumed to exist in the top sheet, emerging on each side of the top rivet hole at its horizontal extremities. The cracks are taken to be straight



and parallel to the x-axis. The energy release rate,  $\mathfrak{S}$ , the mode I and II stress intensity factors,  $K_I$  and  $K_{II}$ , and the compliance change due to the presence of the cracks have been calculated as a function of the crack length for the case of no transverse loading,  $\Sigma_{xx}=0$  for each of the two rivet/sheet boundary conditions.

Figures 3b and 3c give close-ups of the top portion of the outside sheet with the near-tip finite element mesh inserted. Both sheets must be meshed for the crack problem since only the outside sheet contains the cracks, and the problem is no longer symmetric with respect to the two sheets. Refinements of the near-tip mesh have been made for 9 distinct crack lengths. The elements nearest to the tip are composed of quarter-point elements to model the inverse square root singularity of the stresses and strains. The radial length of the smallest elements ranged from  $6.13 \times 10^{-4}R$  for a crack of length  $a/R=0.199$  to  $3.69 \times 10^{-3}R$  for a crack of length  $a/R=3.61$ . The energy release rate,  $\mathfrak{S}$ , was calculated using the J-integral option in ABAQUS. Let  $\psi=\tan^{-1}(K_{II}/K_I)$  be the measure of the relative proportion of mode II to mode I. The values of  $\psi$  were computed using values of the tangential and opening components of the crack face displacements behind the tip. The distance behind the tip used for this evaluation was chosen to be that where the independent evaluation of  $\mathfrak{S}$  from the crack opening displacements gave the best agreement with the more accurate J-integral estimate of  $\mathfrak{S}$ . The values of  $\psi$  are not expected to be as accurate as those for  $\mathfrak{S}$ , but the crack tip is so nearly in a state of pure mode I that this is not a significant consideration.

Define a measure of the magnitude of the stress intensity factors by

$$K = \sqrt{EG} = \sqrt{K_I^2 + K_{II}^2} \quad (4)$$

where the last equality follows from the plane stress relation  $\mathfrak{S}=(K_I^2+K_{II}^2)/E$ . With these definitions

$$K_I = K \cos \psi \quad \text{and} \quad K_{II} = K \sin \psi \quad (5)$$

The curves of  $K/(\Sigma_{yy}\sqrt{R})$  versus  $a/R$  for the two rivet conditions are shown in figure 4, and values are recorded in table 1. These results were computed using the nondimensional lengths in equation (1). The corresponding curves of  $\psi$  as a function of  $a/R$  are given in figure 5. For each of the two rivet boundary conditions the magnitude of  $\psi$  is small; to an extremely good approximation the crack is a mode I crack with  $K_I=K$ . However  $\psi$  is negative for BC#1 implying that there is a tendency for the crack to curve upward slightly, while for BC#2 the trend would be for the crack to curve downward.

There is no hint of an explanation of the "catch-up" phenomenon in the results of  $K$  versus  $a$ , although  $K$  is only a weakly increasing function of the range of  $a/R$  for each of the two cases. If the functional relationship had displayed a local peak for crack lengths on the order of the rivet radius or so, then that feature might help to explain how shorter cracks could



grow faster than somewhat longer cracks and thus tend to catch-up in length. A local peak had seemed a plausible possibility because of the high stress concentration at the edge of the hole, but it does not exist according to either of the present models. It must be remembered that the residual stress contribution to the stress intensity factor has not yet been included in the lower curve in figure 4 for BC#2, and its inclusion can lead to a broad local peak as will be seen in the next section. First, however, an approximate way to include the effect of the transverse loading,  $\Sigma_{xx}$ , on K will be given.

The problem of a traction-free hole with symmetric edge cracks in an infinite sheet has been solved in the literature (refs. 4 and 5) for the case where the remote stress is  $\Sigma_{xx}$ . The result can be written as

$$\frac{K}{\Sigma_{xx}\sqrt{R}} = -\sqrt{\pi\frac{a}{R}}F(s), \quad s = \frac{a}{R+a} \quad (6)$$

where  $F(s)$  is given in the Appendix. For crack lengths which are large compared to  $R$ , this result is clearly a poor approximation because of the interaction of the cracks with the edges of the section. However, the main influence of this contribution is for relatively short cracks. This is evident in figure 4 where  $K$  for the case of no residual stress (BC#1) is given by the superposition of the previous result for  $\Sigma_{xx}=0$  and equation (6) with  $\Sigma_{xx}=\Sigma_{yy}/2$ , corresponding to the proportion of loads for a pressurized cylindrical shell. The influence of the transverse contribution is relatively small over the entire range of  $a/R$ . The accuracy of the simple approximation in equation (6), which neglects any interaction with the rivet, can be readily checked by calculations using the present model. This will be done in future work. Given the small influence of the transverse component of loading on the result in figure 4, it is not expected that more accurate modeling will change conclusions drawn based on calculations carried out with  $\Sigma_{xx}=0$ .

The compliance change due to the presence of the MSD cracks can be computed using the well known relationship between the derivative of compliance with respect to crack length and the energy release rate, or it can be determined directly from the finite element results. As a check on the computations, both determinations were carried out and found to be in agreement.

## THE ROLE OF RESIDUAL STRESSES AT THE RIVET HOLES

For BC#2 it has been assumed that the rivets maintain contact with the holes in the sheets due to residual stress induced in the riveting process. However, the results in figure 4 for this case do not include any effect of the residual stress other than the consequence of full contact. In this section, an approximate analysis of the residual stress contribution to  $K$  will be given. Any contribution to the stress intensity factor due to residual stress at a rivet hole is not cyclic but remains steady as  $\Sigma_{yy}$  and  $\Sigma_{xx}$  undergo cycles of loading and unloading. In this section, we wish

to distinguish between contributions to  $K$  from the residual stress which does not cycle and those from the remote loads which do. For this purpose, now let  $\Sigma_{yy}$  and  $\Sigma_{xx}$  be the *peak* values of the remote loads which are assumed to cycle between zero values and these peak values. Let  $\Delta K$  denote the cyclic change in  $K$  associated with the cyclically applied remote loads, as obtained using the results given in the previous section for BC#2. The estimate for the non-cyclic contribution to  $K$  from the residual stress is then used in combination with  $\Delta K$  to calculate the variation of  $K_{\max}$  and  $K_{\min}$  as a function of  $a/R$  where  $K_{\min}=K_{\text{residual stress}}$  and  $K_{\max}=K_{\text{residual stress}}+\Delta K$ .

An estimate of the stress intensity factor is obtained in the Appendix for cracks emerging from a rivet hole where there is a pre-existing residual stress field due to the riveting process. The model applies to an isolated rivet in an infinite sheet (see the insert in figure 6) and assumes that the radial mismatch strain between the rivet and the hole is  $\epsilon^T$ . The elasticity of the rivet is neglected, and the radial stress exerted by the rivet on the *uncracked* sheet at  $r=R$  is  $\sigma_R=E\epsilon^T/(1+\nu)$ . The stress intensity factor at the tip of the emerging crack with radius  $a$  has the form

$$\frac{K}{\sigma_R \sqrt{R}} = \sqrt{\pi \frac{a}{R}} G\left(\frac{a}{R}\right) \quad (7)$$

This relation is plotted in figure 6.

Curves of the normalized  $K_{\max}$  are plotted as a function of  $a/R$  in figure 7 for various values of the ratio of the residual stress to the cyclically applied remote stress,  $\sigma_R/\Sigma_{yy}$ . Here, the transverse loading,  $\Sigma_{xx}$ , has been taken to be zero, but its effect is not expected to be large, as previously discussed. Companion curves for  $K_{\min}$  are readily generated using the results in figure 6. If one assumes that the riveting process induces some plastic yielding in the sheet, then a reasonable estimate for  $\sigma_R$  should fall between  $\sigma_Y/2$  and  $\sigma_Y$ , where  $\sigma_Y$  is the tensile yield stress of the sheet material. Based on typical values of  $\sigma_Y$  and  $\Sigma_{yy}$  for commercial aircraft, the range of  $\sigma_R/\Sigma_{yy}$  is expected to fall in the range 1.5 to about 3. The curve of normalized  $K_{\max}$  versus  $a/R$  is flat over much of the range of  $a/R$  of interest when  $\sigma_R/\Sigma_{yy}=2$  and actually has a broad peak with its maximum occurring at  $a/R \cong 0.5$  when  $\sigma_R/\Sigma_{yy}=3$ . Any conclusions concerning implications for the "catch-up" phenomenon must also take into account the trends for  $K_{\min}$  and the dependence of the fatigue crack growth rates on both of the parameters characterizing the stress intensity factor cyclic history. Work along these lines will follow in a subsequent paper.

Another feature of the results in figure 7 is that for values of the residual stress parameter  $\sigma_R/\Sigma_{yy}$  in the range from 2 to 3 the values of  $K_{\max}$  are not very different from those which would be predicted for the model with BC#1 in figure 4. When  $a/R$  is larger than about 3 the  $K$ -values for BC#1 reflect the strong interaction between neighboring cracks. This interaction is somewhat



suppressed by the full contact condition which is assumed for the rivet for BC#2. When the cracks become long, the full contact condition is almost certainly unrealistic and separation of the rivet and sheet is expected at the top of the loading cycle. In other words, we believe that BC#1 more realistically models the rivet/sheet interaction even in the presence of residual stress when the cracks become sufficiently long. The present results for the two cases give a clear indication of the range of the stress intensity factor variations which can be expected. It remains to more completely analyze the problem in order to ascertain when one or the other of modeling conditions are appropriate and whether other more detailed models need to be considered.

## LOAD SHEDDING

Another explanation of "catch-up" which seems plausible is connected with load shedding. The load shedding idea suggests that if one crack is longer than its neighbor, it will shed some of the load its section carries to the section of its neighbor, thereby increasing the level of the shorter crack's stress intensity factor and providing a means for the shorter crack to grow faster. By considering alternating long and short cracks of length  $a_1$  and  $a_2$ , as indicated in figure 8, one can develop a model along the lines of that described in the previous two sections except that now the periodic section is twice as wide. The section used in carrying out the finite element calculations is shown in figure 8. Precisely the same element meshes used in the earlier sections were used here, but these were now linked together to provide meshes appropriate for the two differing crack lengths. Table 2 lists the stress intensity factors calculated for the unequal crack lengths for the case of rivet conditions BC#1. Included in this table are the stress intensity factors for each of the crack lengths as calculated in the previous section where all cracks are of equal lengths. The differences between the stress intensity factors for a crack of given length in the two calculations is less than 1 percent. One concludes from this that, for cracks with  $a/R < 1.5$ , there is almost no crack interaction and negligible load shedding.

## CONCLUSIONS

Basic results have been presented for stress intensity factors for MSD cracks. These results bring out the sensitivity of the predictions to the assumptions made in modeling the rivet/skin interaction. Further work, both theoretical and experimental, needs to be carried out to better characterize this interaction. The results for the two modeling assumptions made in this paper are not substantially different when the residual stress contributions to the stress intensity factors are taken into account. Nevertheless, these predictions should be regarded as preliminary subject to validation by further study. In addition, no effort has been made to include effects due to out-of-plane bending, plasticity, or three dimensionality. One or more of these effects may be



important in MSD cracking. Finally, no dramatic indicator of a catch-up mechanism has emerged in the present work, although the predicted stress intensity factor variation with crack length is relatively weak, especially for the case involving residual stress, suggesting that long and short cracks would grow at roughly the same rate.

### ACKNOWLEDGEMENT

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### APPENDIX

#### *Stress Intensity Factor due to Transverse Loading*

The result in equation 6 for the transverse loading  $\Sigma_{xx}$  with  $\Sigma_{yy}=0$  has been taken from reference 5. The expression for  $F(s)$  is

$$F(s) = 0.5(3-s)\left[1 + 1.243(1-s)^3\right] - 1 - (1-s)\left[0.5 + 0.743(1-s)^2\right] \quad (A1)$$

### *Stress Intensity Factor due to Residual Stress at the Rivet Hole*

This is an approximate result which is developed using the solution for the stress intensity factor given in reference 5 for the problem of a hole of radius  $R$  with cracks of length  $a$  emanating from its horizontal extremities, where the surface of the hole (but not the crack faces) is subject to a uniform pressure,  $p$ . That solution is  $K=p(\pi a)^{1/2}H(s)$  where  $s=a/(R+a)$  and

$$H(s) = (1-s) \left[ 0.637 + 0.485(1-s)^2 + 0.4s^2(1-s) \right] \quad (A2)$$

The approximate solution will also make use of the *additional* average outward radial displacement at the hole surface  $\delta$  due to the presence of the two edge cracks of length  $a$ . This quantity can be derived from the relation between  $\delta$  and the energy release rate

$$G = -\frac{1}{2} \left( \frac{\partial(\text{Potential Energy})}{\partial a} \right)_p = \frac{\pi}{2} R p \frac{\partial \delta}{\partial a} \quad (A3)$$

The result is  $\delta=(2pR/E)f_\delta(a/R)$  where

$$f_\delta \left( \frac{a}{R} \right) = \int_0^{a/R} H(s)^2 \frac{a}{R} \frac{da}{R} \quad (A4)$$

The radial displacement of the hole without cracks present when subject to a pressure  $p$  is  $u=pR(1+\nu)/E$ . If it is assumed that there is a strain misfit between the rivet and the hole,  $\epsilon^T$ , and that the rivet is effectively rigid, then an approximate condition for determining  $p$  as a function of the length of the cracks is  $u+\delta=R\epsilon^T$ . This equation gives

$$p = \sigma_R \left[ 1 + \left( \frac{2}{1+\nu} \right) f_\delta \left( \frac{a}{R} \right) \right]^{-1} \quad \text{where} \quad \sigma_R = \frac{E\epsilon^T}{1+\nu} \quad (A5)$$

Note that  $\sigma_R$  is the normal stress exerted by the rivet on the hole when  $a=0$ . This result in combination with equation A2 gives the desired result, equation 7, where

$$G \left( \frac{a}{R} \right) = H(s) \left[ 1 + \left( \frac{2}{1+\nu} \right) f_\delta \left( \frac{a}{R} \right) \right]^{-1} \quad (A6)$$

Table 1

Normalized Stress Intensity Factors  $K_{\text{norm}} = \frac{K}{\Sigma_{yy} \sqrt{R}}$  for  
Two Sets of Rivet/Skin Boundary Conditions ( $\Sigma_{xx}=0$ )

<u>a/R</u>	<u>K<sub>norm</sub> for BC#1</u>	<u>K<sub>norm</sub> for BC#2</u>
0.199	2.440	0.502
0.371	2.723	0.656
0.586	2.906	0.799
0.855	3.072	0.948
1.192	3.268	1.109
1.612	3.530	1.291
2.137	3.899	1.502
2.794	4.479	1.763
3.615	5.557	2.125

Table 2

Results for the Load Shedding Analysis  
Using BC#1 Rivet/Skin Interaction Conditions ( $\Sigma_{xx}=0$ )

<u>a/R for Crack 1</u>	<u>a/R for Crack 2</u>	<u>K<sub>norm</sub> for Crack 1</u>	<u>K<sub>norm</sub> for Crack 2</u>
0.586	0.586	2.906	2.906
0.855	0.586	3.069	2.908
0.855	0.855	3.072	3.072
1.192	0.855	3.265	3.075
1.192	1.192	3.268	3.268



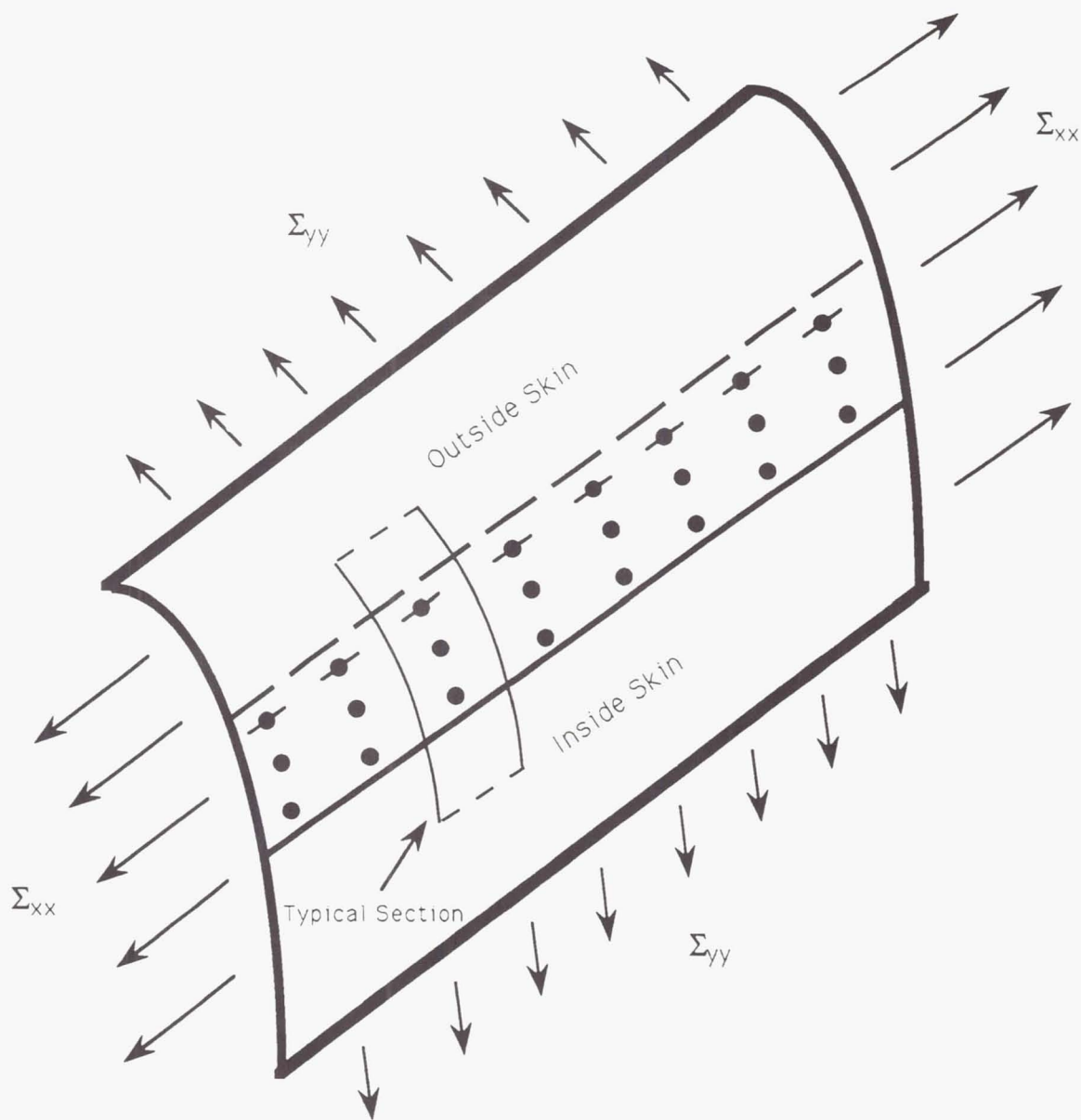


Figure 1 Geometry and loading of the lap joint.

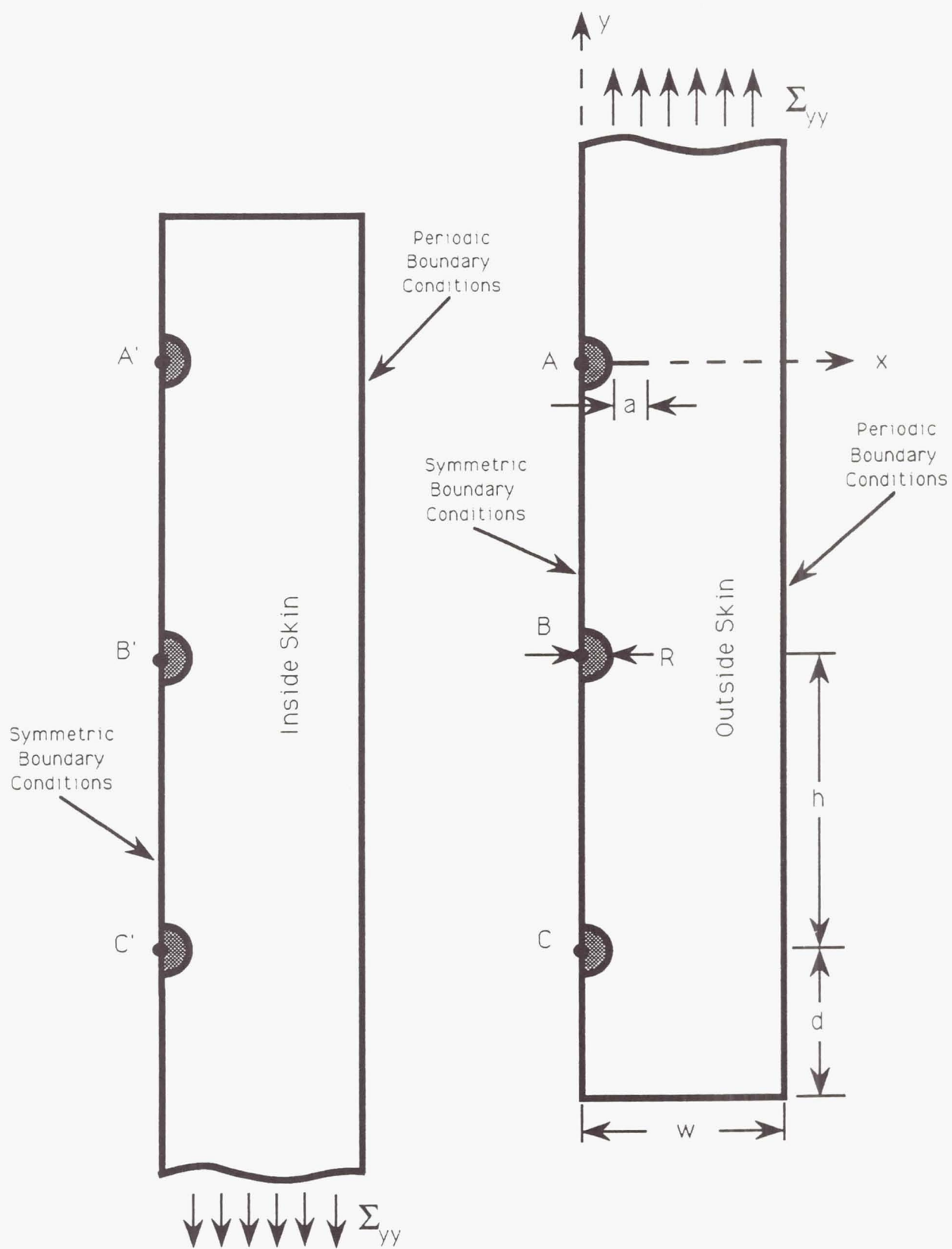


Figure 2 Sections used in the finite element model with  $\Sigma_{xx}=0$ .

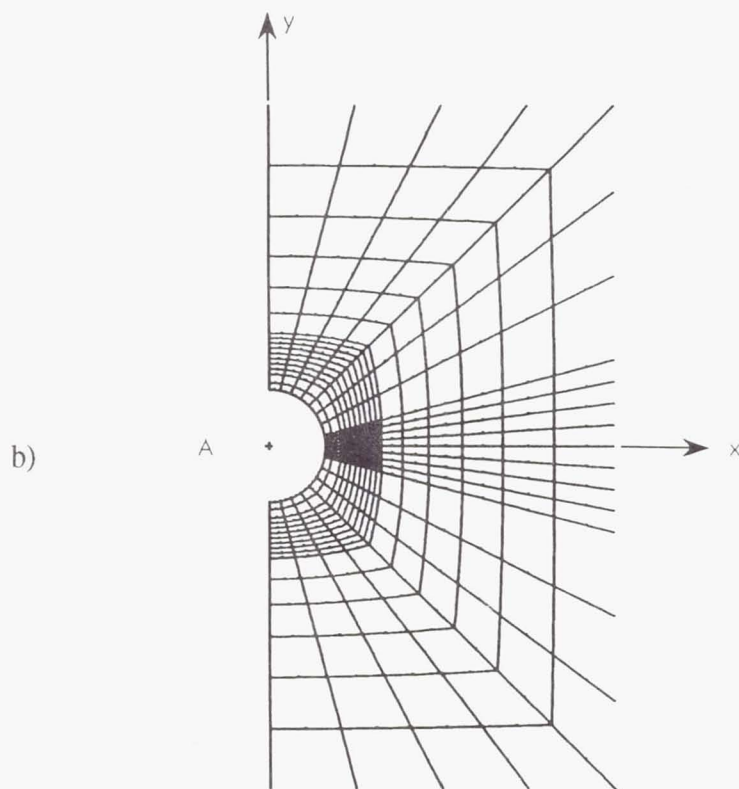
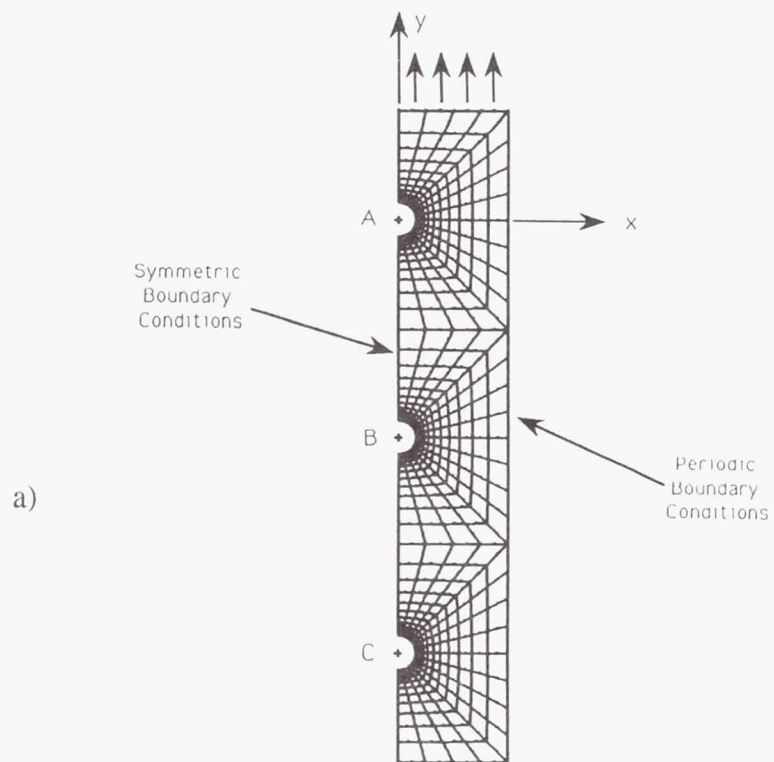


Figure 3 a) Finite element mesh for section without cracks. b) Local mesh near the cracked rivet hole.



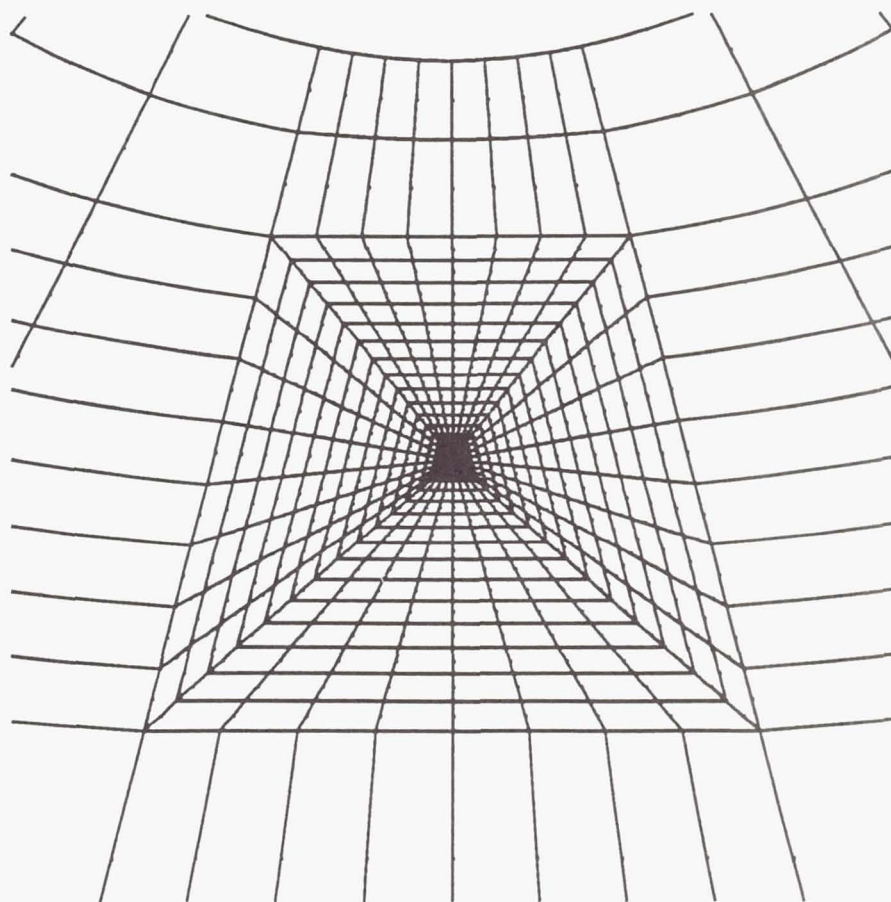


Figure 3c) Close-up of local mesh near the cracked rivet hole.

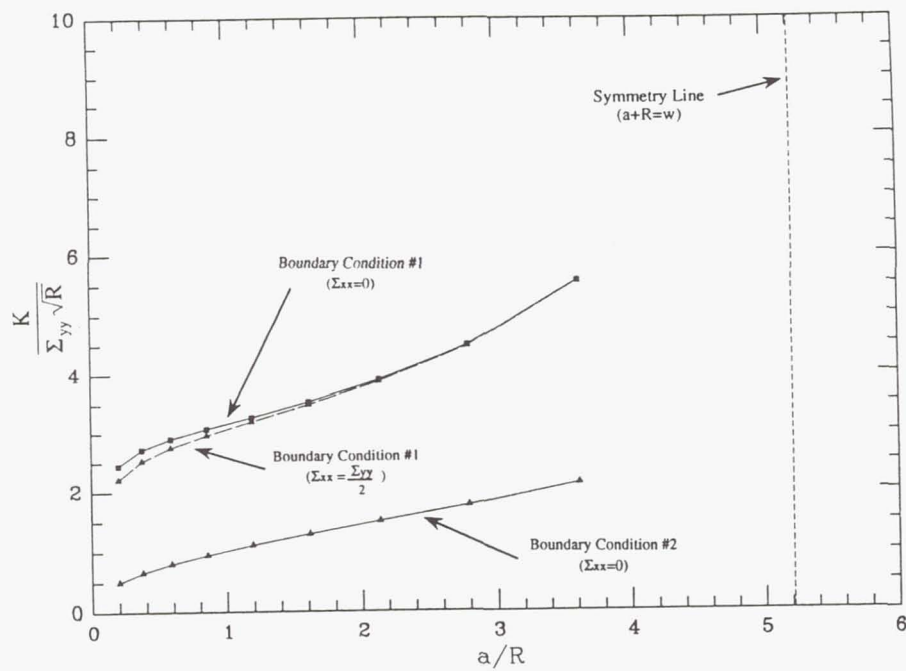


Figure 4 Normalized stress intensity factors as a function of normalized crack length for the two sets of rivet/skin boundary conditions.

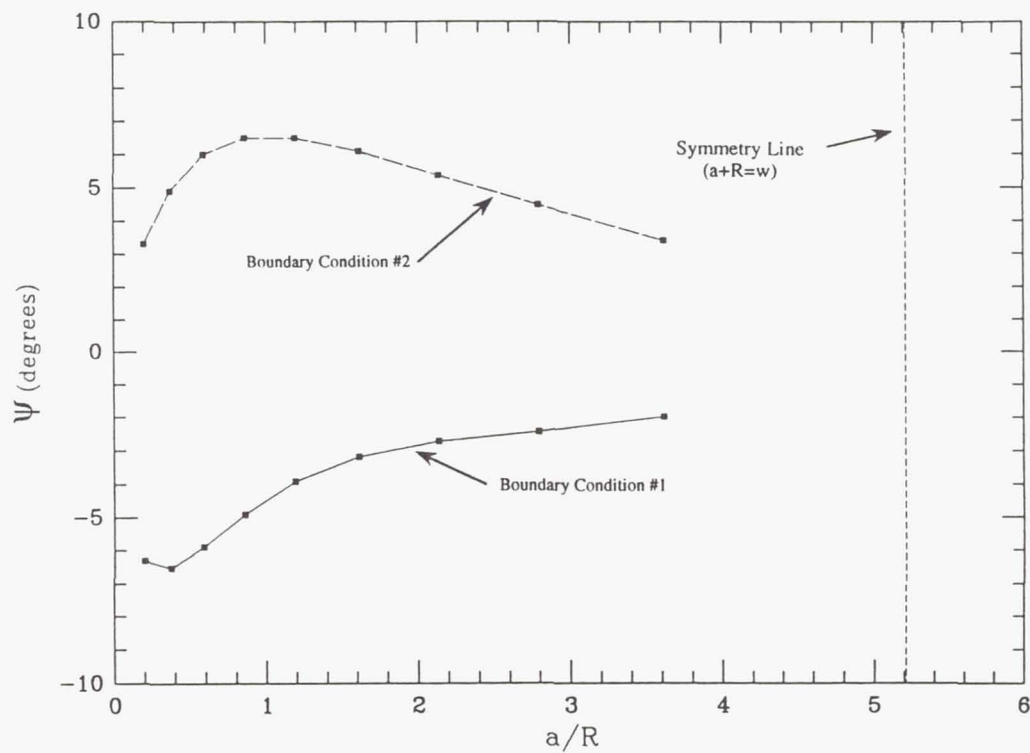


Figure 5 Measure of mode mixity at the crack tip for two sets of boundary conditions.

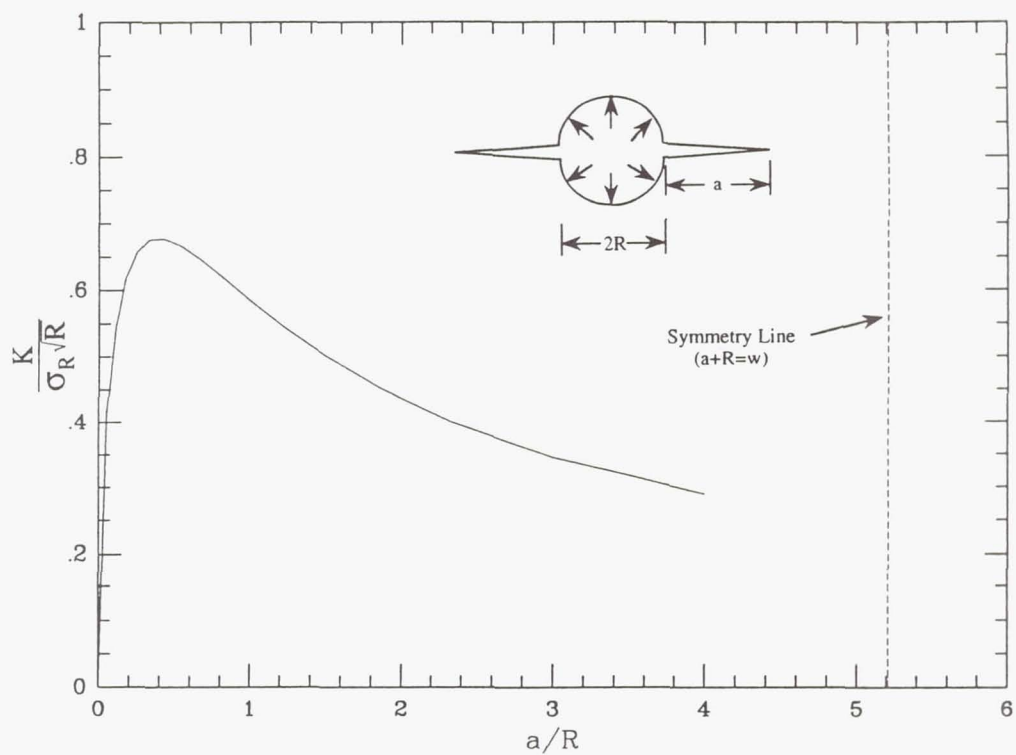


Figure 6 Normalized stress intensity factor contribution due to residual stress.

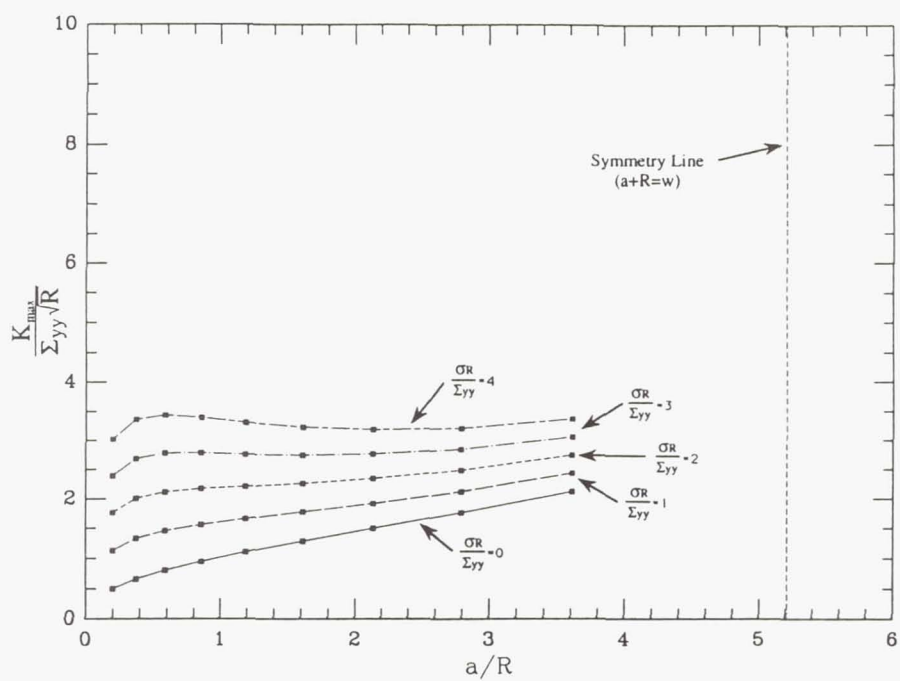


Figure 7 Normalized maximum stress intensity factors for Boundary Condition #2, including the effect of residual stress.



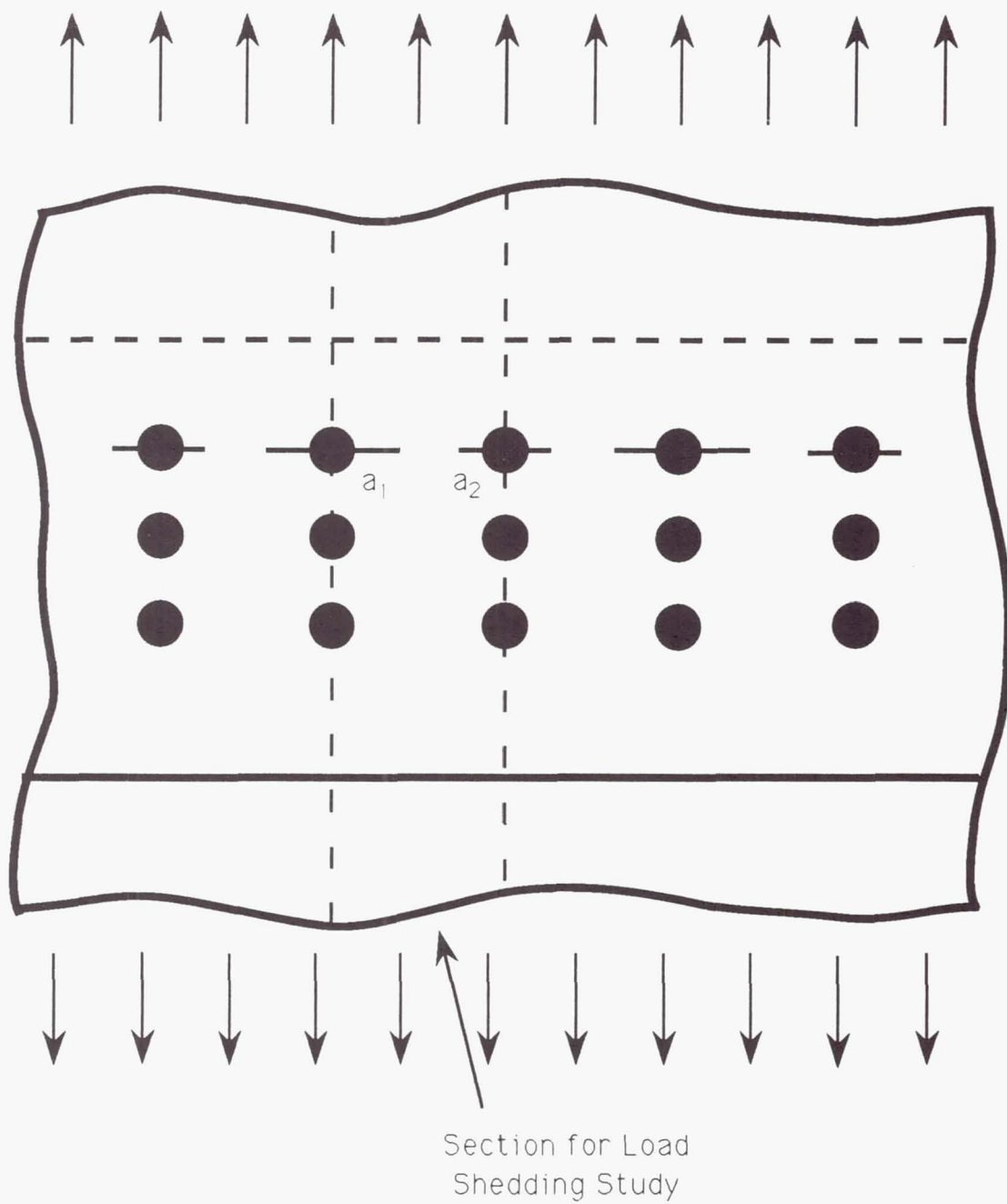


Figure 8 Geometry for the load shedding study.

CURRENT DOT RESEARCH ON THE EFFECT  
OF MULTIPLE SITE DAMAGE ON STRUCTURAL INTEGRITY

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ABSTRACT

Multiple site damage (MSD) is a type of cracking that may be found in aging airplanes and which may adversely affect their continuing airworthiness. The Volpe National Transportation Systems Center has supported the Federal Aviation Administration Technical Center on structural integrity research for the past two and half years. The work has focused on understanding the behavior of MSD, detection of MSD during airframe inspection, and the avoidance of MSD in future designs. This paper addresses these three elements of the MSD problem and provides a summary of the work done, the current status, and requirements for future research.

INTRODUCTION

The occurrence of multiple site damage (MSD) in older aircraft was highlighted by the in-flight failure of a portion of the fuselage of an Aloha Airlines Boeing 737 in April 1988. The failure was precipitated by the linkup of small fatigue cracks emanating from adjacent rivet holes in the lap joint of the fuselage. Special inspections following the incident have detected MSD in aging aircraft of different makes and models. Retrospective consideration of the 1983 failure of the JAL Boeing 747 aft pressure bulkhead and the widespread cracking in the wings of the original Air Force KC-135 and C-5A suggests that MSD can occur at locations other than the fuselage. Thus, it is prudent to assume that MSD may have the potential to appear anywhere in the nation's older fleets.

This has raised concern about the continuing airworthiness of older aircraft. After the incident, the private sector and the Federal Aviation Administration (FAA) quickly went into

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action to maintain the structural integrity of older transport category airframes that are currently in service. At the International Conference on Aging Airplanes in June 1988, several interrelated technical areas were identified as key to the proper understanding of the aging airplane problem [1]. The FAA has since developed a National Aging Airplanes Research Program [2] to determine if current rules for design, inspection, and maintenance are sufficient to ensure the safe operation of the aging fleet. At the Second International Conference on Aging Airplanes in October 1989, progress of research efforts and actions taken by the FAA and the industry were reported [3]. In a subsequent meeting [4], many of the technical issues on structural integrity were addressed.

The Volpe National Transportation Systems Center (VNTSC) has supported the FAA Technical Center since February 1989 to implement the structural integrity portion of the FAA program which focuses on research to understand the behavior of MSD, techniques to find MSD during airframe inspection, and countermeasures to avoid MSD in future designs<sup>2</sup>. This paper addresses these three elements of the MSD problem in structural integrity research and provides a summary of accomplishments and future plans.

#### UNDERSTANDING MSD BEHAVIOR

As in the case of tolerance of isolated cracking, tolerance of MSD can be described in terms of detectable crack size, critical crack size, and the number of flights or flight hours of slow crack growth between these limits. All three factors can have values for MSD which are quite different from the range of values generally associated with isolated cracking. Several parallel research efforts are in progress to identify these factors for MSD.

#### Definition of Multiple Site Damage

Multiple site damage (MSD) is generally characterized by a group of cracks originating from similar structural details located in a common area. MSD may consist of small cracks of similar size or a large lead crack growing toward a group of small cracks. In order to define MSD more precisely, a three-part definition was proposed [7] as follows:

1. MSD is the occurrence of independent cracks which may linkup to cause greater damage or partial fracture, for which the residual strength is less than it would be if such greater damage developed as a consequence of growth of a single crack.

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<sup>2</sup> The earlier progress of this work was reported in [5,6].



2. This linkup may become unstoppable due to further interaction with otherwise subcritical cracks ahead of the partial fracture.
3. A change in inspection procedure or interval from that for a single crack is needed.

Under the present definition, the occurrence of multiple cracks is not MSD if no linkup occurs or the linkup of these cracks does not result in the reduction of residual strength below the limit load which defines plastic collapse.

An example of true MSD is shown in Figures 1 and 2. The cracks are initially independent. Many can initiate and progress almost simultaneously. The linkup of these cracks may not cause catastrophic failure if the structure can sustain a two-bay skin crack at the limit load and no further cracks exist beyond the affected two bays (this would be a normally arrested fracture). However, if further cracks beyond the large one linkup to join the main fracture at low stress, the fracture may proceed uncontrollably and lead to the zipper effect. The phenomenon of cracks linking up is not specifically addressed in the current damage tolerance requirements for transport category airplanes [8,9]. Those requirements consider only the isolated cracks which constituted the airframe service fatigue experience base up to the mid-'70s.

Figure 3 shows schematically the residual strength diagram for a skin-stringer (strap or frame) combination. The structure is normally designed such that the residual strength at point A coincides with or is above the limit load. Then fracture, resulting from one of multiple cracks, will be arrested upon reaching the adjacent stiffening element because of load transfer to this element.

Now consider the MSD case of Figure 1. Its residual strength diagram is shown schematically in Figure 3 with and without MSD. The first linkups occur at relatively low stresses. Due to MSD ahead of this fracture, point A is reduced to point B in Figure 3. In other words, for stresses above point B the structure has lost its arrest capability, and the fracture can proceed at stresses below the limit load for which it was designed. Another consequence is that the critical crack length  $a_p$  is also reduced.

MSD as defined may not develop extensively in all aging aircraft. Multiple cracks might, but they are covered by damage tolerance analysis and tests. The condition for the occurrence of true MSD is the presence of sites of similar configuration subject to almost the same stress. Such cases may be few.

Chordwise stress gradients in a wingbox dictate that there will be few cracks in close proximity, which are covered by the

arrest capability at limit load. MSD is more likely in longitudinal fuselage fastener rows. At other locations, including circumferential splices, stress gradients are such that the occurrence of true MSD not covered by the damage tolerance analysis is less likely. For these configurations, a few adjacent cracks may develop. Nonetheless, after their linkup they will form a single crack which was considered in the damage tolerance analysis.

### Tear Strap Effectiveness

Current design practices include the use of frame members and, in certain instances, tear straps, which act with other stiffening elements to allow the fuselage to withstand an isolated longitudinal crack up to two bays long at 110 percent of normal operating pressure plus the aerodynamic pressures in lg flight. Fail-safety is achieved through "flapping," in which a long crack changes direction causing a controlled decompression.

One objective of the current research is to determine how effectively frames and tear straps can contain fracture in the presence of MSD. The general scenario assumes a fracture resulting from linkup of a group of MSD cracks. The fracture in this case lies along a skin splice, rather than in the mid-bay position usually assumed in present design and test practices. Also, the fracture may be advancing toward adjacent bays which contain additional (but as yet unlinked) MSD cracks.

A special fixture, which is shown in Figure 4, has been constructed to examine the residual strength and fatigue life of curved panels 68 inches on the circumference by 120 inches axially, with allowance for a range of 70 to 75 inches in radius [10].

The fixture is a shallow pressure box which accommodates the test article by means of floating seals. Hydraulically applied pressure produces hoop stress in the skin, which is reacted through lateral turnbuckles. Hydraulic cylinders and turnbuckles at the ends of the panel produce an axial stress proportional to and in phase with the hoop stress, thereby simulating the biaxial stress state found in a fuselage.

The test panels are reinforced by frames and tear straps in the circumferential direction and by stringers in the longitudinal direction as shown in Figure 5. The dimensions, construction details, and materials were chosen to replicate a configuration similar to those found in aging airplanes. The test panels do not precisely match the design of any actual aircraft model, and therefore, the test results are intended to have only generic interpretations. However, the panels will provide stress levels and structural flexibility which lie in the range of existing designs. Test results will thus be sufficiently realistic for the purposes of drawing general



conclusions about MSD behavior and calibrating damage tolerance estimation procedures.

A series of shakedown tests were performed [10] to demonstrate the successful operation of the test rig and to compare the strain fields in the panel to those calculated for a full fuselage. The initial checks were successful. The tests included five residual strength tests, where the test panels were subjected to an increasing pressure load until unstable crack growth was observed, and one fatigue test. The results of the residual strength tests are shown in Figure 6.

The effectiveness of tear straps in arresting or containing a moving crack was examined. In all of the shakedown tests involving mid-bay cracks, once instability was reached, the cracks turned at an angle of nearly  $90^\circ$  at the next tear strap, thereby containing or arresting the crack through flapping. Flapping, however, did not occur in the two tests involving lap splice cracks; one with and the other without MSD. That is, the lap splice cracks extended in the same direction as the crack, or in a self-similar manner. However, the early results from the second series of tests, that are presently ongoing, indicate that flapping may occur even for lap splice cracks. These most recent test results will be reported on in the future.

In the fatigue test, intentional damage was built into a panel in the form of MSD-type cracks along the upper row of rivets on the outer skin of the lap joint. In order to accelerate the crack growth process, the maximum pressure applied to the panel during cycling was 1.5 times the normal operating pressure for the last 18,300 pressurization cycles. After a total of 68,340 cycles, the panel was removed from the test fixture due to failure from cracks that had formed on the lower row of rivets on the inner skin of the lap joint. Fractographic analyses were initiated to explain this unexpected failure since the upper row, not the lower row, was notched. The results of the fractography [11] suggest that local bending amplified by the overpressure may have been the cause of the lower row failure. A striation spacing analysis of a rivet fracture surface determined that 23,500 cycles occurred between a point near one of the crack origins and a point near the switch from fatigue to ductile tearing. Another observation from the fractography is that the cracking in the tear straps occurred at the same time MSD was developing. This may further degrade the fail-safe capability of the structure.

In parallel with the curved panel test effort, analytical models have been developed to predict the failure pressure of the tests. One method of analysis, generally accepted by the aircraft industry, is the compatible displacement method [12, 13] which can readily account for the effects of rivet flexibility, biaxially applied stress, and stiffener bending. Analyses were performed assuming linear rivet flexibility (as measured in separate experiments). Comparisons between analytical and



experimental results are shown in Figure 7 for mid-bay cracks. As shown, the analytical results are dependent on the assumed value for the critical stress intensity factor. The experimental results agree well with the analytical results for  $K_{Ic}$  between 150 to 160 ksi $\sqrt{\text{in}}$ . The compatible displacement analysis can be easily modified to account for the effects of nonlinear or piecewise linear rivet flexibility by implementing an iterative procedure. Figure 8 shows the analytical results from a compatible displacement analysis assuming nonlinear rivet flexibility.

Additional tests are currently being conducted that will demonstrate the residual strength of curved panels containing cracks in the lap splice. A preliminary test matrix has been developed that includes testing three (3) crack lengths with and without MSD. Also, additional fatigue tests will be conducted that will determine multiple site crack growth behavior and the effectiveness of the terminating action on curved fuselage panels.

#### Causes and Likelihood of MSD

To identify and counter MSD as it appears in the current fleet, and to avoid MSD in future designs, it is important to understand which design features have an increased susceptibility to develop MSD, and to predict how likely MSD is to occur in a given design.

A test program is underway to identify design features which have higher MSD potential [14]. The panels selected for fatigue tests are flat 12-inch wide panels, made from 2024-T3 clad aluminum, which contain a lap joint with three or five rows of rivets, and are reinforced along the edges to simulate stiffeners in a fuselage. A strip of aluminum is attached to the middle row of rivets to account for the local change in thickness from a stringer that is usually attached at that location on an actual aircraft. The strip has the same thickness as the crown of the stringer (nominally, 0.04 inches). A schematic of the lap joint specimens used in these tests is shown in Figure 9. The design was chosen on the basis that the panel is simple to construct and gives a stress distribution adequately representative of that in a fuselage. Figure 10 compares the stress distribution of the flat 12-inch panel with the stress fields in a fuselage as calculated by the finite element method. Additionally, MSD occurred in these panels at a life reasonably consistent with what has been observed in service for some aircraft. Figure 11 compares the crack patterns from a test on a 12 inch wide panel with those from an actual aging airplane at approximately the same number of cycles.

The test program is divided into four series. The loading in most of the tests is uniaxial tension, but some tests apply

combined tension and shear to determine the effect of mixed-mode loading on fatigue crack growth.

The first series was conducted to determine the effect of the terminating action which was mandated by the FAA for certain aging aircraft with MSD-prone lap joints. The terminating action replaces the upper-row countersink rivets with larger, button-head rivets. In particular, these tests investigate the concern that, following the terminating action, MSD may recur in the inner skin on the lower rivet row where it is difficult to detect. Referring to Figure 12, the results show that the terminating action is effective in reducing the growth rate of cracks emanating from rivet holes in the upper row of rivets.

The effect of shear loading on fatigue life of the panels was examined in the second series. These tests were conducted by applying the remote loading at an angle, thereby inducing a shear load in addition to the applied tensile load. Figure 13 shows a schematic of the load application set-up. Photographs of MSD-type cracks emanating from rivet holes at an angle of approximately  $20^\circ$  are evidence that a shear stress component exists in actual aircraft fuselages. For a shear-to-tension load ratio of 0.1, a reduction in fatigue life by a factor of 1.5 was observed in these tests.

The third series was designed and carried out to determine which of the following parameters have a significant influence on fatigue life and MSD formation:

<u>Parameter</u>	<u>Levels Tested</u>
a. Stress level	12, 14, and 16 ksi
b. Rivet type	flush head and Briles
c. Rivet spacing	0.75, 1.00, and 1.29 inches
d. Rivet orientation	continuous and staggered
e. Number of rivet rows	3 and 5
f. Skin Thickness	0.04, 0.05, and 0.063 inches

Three parameters with three levels and three parameters at two levels for a total of  $216 (3^3 \times 2^3)$  unique combinations were possible. However, rather than establishing a baseline set of design parameters and varying each parameter one at a time, a statistical theory was used to design this experiment to reduce the number of unique test configurations while assuring that sufficient meaningful inferences could be made from the test results. A fractional factorial plan [15] was developed which reduced the number of configurations from 216 to 27; these are listed in Table 1. This experimental design also allows for evaluating the significance of pre-determined interactions. That is, the interactions of stress with rivet spacing, stress with skin thickness, and rivet spacing with skin thickness can also be evaluated for significance to fatigue life and MSD formation.

The completed data set of cycles-to-failure for all 27 configurations was analyzed by the variance method as



implemented through the Statistical Analysis Software (SAS) computer program, a commercially available software package. This program was used to test for significant differences in cycles-to-failure that could be uniquely attributable to each of the six parameters and the three interaction terms estimated from design. The results of the statistical analysis are shown in Table 2 where any parameter with a corresponding F value less than 0.05 is considered to be significant. That is, stress level, rivet type, and skin thickness have a statistically significant influence on the number of cycles to failure. Further analysis showed that the effect of stress level and rivet type is as expected; lower stress levels increase the fatigue life and the Briles rivet, which eliminates the knife edge, and also increases the fatigue life. On the other hand, the effect of skin thickness is not intuitively obvious. The preliminary analysis indicates that the greatest fatigue life is obtained with the smallest skin thickness, even though the knife edge is sharpest in this configuration. The effect of bending in the lap joint may be more significant for thicker skins.

Additional tests will be conducted to substantiate these results, and to evaluate the effect of bending on fatigue life. The statistical analysis will be repeated to determine the parameters that have a significant effect on the formation of multiple site damage.

### Corrosion and Structural Integrity

Future experimental work on lap joint specimens will be conducted to determine the effect of pre-existing corrosion on fatigue initiation and crack growth behavior. A corrosion protocol will be developed that will replicate the different forms of corrosion in an accelerated manner. The ASTM standards ASTM G34 and ASTM G44 are being considered as possible candidates for the protocol. ASTM G34, also known as the "EXCO" test, is a total immersion test that replicates exfoliation corrosion by using a solution of sodium chloride, potassium nitrate, and nitric acid. ASTM G44 is an alternate immersion test that replicates pitting and intergranular corrosion, and uses a solution of sodium chloride (3.5% by weight) and water. In addition, a seacoast environment will be used as a control on these protocol experiments. Mini-lap joint specimens have been specially made of unclad 2024-T3 aluminum and will be immersed into each solution for varying times of exposure. Once an appropriate protocol has been developed, lap joint specimens will be immersed into the solution and subsequently fatigue tested. The parameters that will be varied in the fatigue testing of pre-corroded panels are: stress level, heat treatment, and degree of corrosion in terms of exposure time. Heat treatment is a variable because aluminum manufactured twenty years ago was quenched at a slower rate than aluminum manufactured today. The



tests conducted under work related to "Causes and Likelihood of MSD" will be used as the baseline for uncorroded specimens.

### Improved Analysis Methods

Fracture stability analyses of stiffened flat panels have been used to correlate test data and predict damage tolerance for over two decades [12]. The models most widely used for these analyses are based on the displacement compatibility method [12,13], in which the effects of load transfer through fasteners are represented by displacement influence functions in the skins and stiffeners. Recent work has shown that the hybrid finite element method can provide excellent results more efficiently by incorporating the skin stress influence functions directly in the element [16]. The hybrid method is versatile enough to take into account the effects of complex geometry variations and different fastener configurations. It can also account for the stress singularities at crack tips and rapid stress variations near rivet holes.

The hybrid finite element method is based on a variational formulation in which relevant field variables in the element need not satisfy a priori the requirements of interelement displacement compatibility and interelement traction reciprocity. The constraint conditions can then be included in the functional by the use of Lagrange multipliers, which are the additional variables at the element boundary. The method can provide directly the solution for the strengths of singularities (such as stress intensity factors at the tips of a crack). Hybrid elements have been extensively developed and used and have been shown to be extremely accurate and efficient in comparison to the standard finite element method [16-20].

A typical problem encountered in damage tolerance analysis involves a panel with multiple cracks. As shown in Figure 14, a cracked panel is connected to stiffeners by rivets and loaded by in-plane remote stresses. The panel also is subjected to concentrated loads along rivet hole surfaces. The entire domain for the problem can be treated as a single element, commonly called a "super element".

The hybrid variational functional for the element can be written as

$$\begin{aligned} \pi_p = t \left[ \int_{\partial A} \left( T_i \tilde{u}_i - \frac{1}{2} T_i u_i \right) ds + \sum_k \int_{|z - z_k| = \epsilon} \left( T_i \tilde{u}_i - \frac{1}{2} T_i u_i \right) ds \right. \\ \left. + \sum_k \int_{\Gamma_k} \frac{1}{2} T_i u_i ds - \int_{\partial A_0} T_i^0 \tilde{u}_i ds \right] \end{aligned}$$

where  $t$  is the panel thickness,  $\partial A$  is the boundary of the element,  $\tilde{u}_i$  are the displacements at the element boundaries. These displacements are in common with those of adjacent elements at nodal points and are piecewise polynomials along the element

boundaries.  $T_i (= \sigma_{ij} v_j)$  and  $\sigma_{ij}$  are respectively the boundary tractions and stresses,  $T_i^o$  are the prescribed tractions on  $\partial A_o$ . The surfaces of the rivet holes are located at  $|z - z_k| = \epsilon_k$ , and the crack surfaces are located along  $\Gamma_k$ . The quantities  $\bar{u}_i$  and  $u_i$  are separate displacement fields which are independent functional variables of  $\pi_p$ .

The stiffness matrix for this element is then obtained by integrating over the crack surfaces and along the rivet hole surfaces. The stiffeners are modeled as beam elements. The rivets are modeled as shear springs. The energy for the beam elements and shear elements are easily expressed in the usual way. The independent variables for the problem are (i) rivet forces at rivet holes, (ii) fracture parameters representing crack singularity, (iii) stiffener deflections. The complex variable theory of elasticity is used in the formulation with Chebyshev polynomials used to represent the stresses at crack locations [20].

As a special case of the above formulation, the problem of a stiffened panel with a single crack and a broken center stiffener was analyzed. The distance between stiffeners is 12 inches and the crack length is  $2a$ . Table 3 compares the present results (MSD) with those of the displacement compatibility method of Swift [13] and the previous finite element work of Tong [16]. Excellent agreement is seen for the correction factor  $\beta$  associated with the stress intensity factor.

The mode 1 stress intensity factors for the problem involving two cracks, Figure 14, are shown in Table 4. The two central stiffeners are assumed to be broken at the crack positions, while the two outer stiffeners are intact. The distance between stiffeners is 12 inches, each crack is of length  $2a$  and the distance between cracks is denoted by  $d$ . Table 4 shows that the stress intensity factor at the outer tip increases slowly with crack length, while that for the inner tip increases much faster as the cracks approach each other. Additional information for this problem is shown in Figure 15 for the residual strength of the structures with a panel fracture toughness of  $K_{Ic} = 120 \text{ ksi}\sqrt{\text{in}}$  and a stiffener ultimate strength of 82 ksi. When the crack tip is about half way to the outer intact stiffener, the stiffener residual strength is only reduced by 14.66% but the panel residual strength (based on the outer crack tip) is reduced by 72.4%.

A typical MSD situation is the lap joint shown in Figure 16, in which the cracks emanate from rivet holes in the skin. Therefore the rivet forces on the hole will have a significant contribution on the fracture parameters for the panel. To solve this problem the hybrid finite element method described earlier is used in conjunction with solutions for plates with cuts subjected to a concentrated load at the cut surface. Stress intensity factors at the crack tips can be calculated. This work is currently being developed and should prove useful in



understanding the mechanism of multiple site damage at lap joints.

The ability to model the curvature and bulging effects of fuselage panels also requires further research to make the extension to MSD. It is well established, through the results of comparative tests, that curved panels behave differently from flat panels, and that the curvature also interacts with cracks to affect properties such as damage tolerance. Flat panel models must, therefore, be empirically calibrated by comparison with curved panel tests in order to predict damage tolerance for fuselage structure. Flat panel model correction factors have been developed for the prediction of tolerance to isolated damage, but MSD correction factors have not been developed. The research approach in this case is to develop better analysis methods which account for curvature effects based on established principles of mechanics and thus require fewer validation tests than the purely empirical approach. Analysis methods are being explored or under development for determining stress intensity factors for MSD cracks in stiffened fuselage panels. The methods will account for the effects of yielding and bulging near the crack tips. These methods will be discussed in detail elsewhere in this conference.

#### Basic Fracture Resistance

A better understanding of basic fracture resistance properties is required in order to predict the tolerance of panels to MSD. Panel fracture predictions are currently made by means of R-curve methods, which account for the fact that a large isolated crack in a thin ductile skin can undergo stable extension at stress levels below the fracture strength. The data from which material R-curves are derived come from tests of center-cracked panels or compact tension specimens with either pin or wedge loading [21]. In all cases the specimen dimensions and length of the single crack are large if the material under test has appreciable ductility.

The results of such tests may not be applicable to damage tolerance assessments in the presence of MSD. Calculations based on the Aloha post-accident observations suggest that conventional R-curves give unconservative estimates for critical crack length. With assumptions of 1.15-inch rivet pitch and skin bypass stress corresponding to design maximum pressure, a critical crack length of 1.1 inches (tip-to-tip) is predicted from a 2024-T3 R-curve, whereas a retired 737 airframe tested by Boeing appears to suggest that the critical crack length is only 0.66 inches. Thus, a need exists to obtain coupon type data which characterizes material fracture resistance in the presence of joining details with which the MSD is associated. In fact, it is by no means clear that the R-curve approach is applicable to MSD linkup, which may be controlled by local plastic collapse (net section type failure).



As a consequence, experiments were initiated to derive special R-curves and to define the fracture resistance of panels with multiple site damage. These experiments were performed on coupons made of 2024-T3 clad aluminum sheet, 0.04 inch thick and 4 or 8 inches wide, with one row of three open holes of 0.1875-inch diameter at 1-inch pitch on center and collinear saw cuts simulating MSD [10]. Figure 17 shows a schematic of the coupon configuration. Coupons with crack lengths of 0.15, 0.22, and 0.26 inches (3.8, 5.6, and 6.6 mm) were tested. Crack growth was monitored as the coupons were subjected to increasing load until linkup. The data revealed that stable crack extension occurred even in the regime where the net section yield criterion would have predicted plastic collapse. There appeared to be no significant difference in the behavior of specimens from the as-fabricated state (blunt saw cuts) and after pre-cycling at low load to initiate a sharp fatigue crack from the central notches.

The test data seemed to suggest that the R-curve for the simulated MSD depended on initial crack length as well as extension, i.e., there was apparently no "master" curve independent of initial crack length, such as was found in conventional R-curve tests. The strain energy density criterion was used in conjunction with an elastic-plastic finite element analysis to predict the stable crack growth that was observed in these coupon tests [22]. Referring to Figure 18, reasonable, but not precise, agreement was obtained from these analyses. Differences between the experimental data and theoretical predictions can be attributed to the approximate procedure used in the elastic-plastic finite element calculations, or to the degree of accuracy in visually measuring the crack extension through a 20X microscope. Notwithstanding the lack of better agreement, these results show that the strain energy density approach provides a feasible criterion that can be used for damage tolerance evaluation of MSD-type cracks where the stability limit may be either controlled by plastic collapse or ductile fracture. An additional merit of the strain energy density criterion, not demonstrated in the present work, is its applicability to mixed mode crack growth conditions under which MSD-type cracks are commonly found.

Even if the R-curve approach is ultimately found to be inapplicable to MSD, it is still useful to have load versus crack extension data for the purpose of calibrating a plastic collapse model. However, observing crack extension through the optical microscope has thus far proved to be a difficult task because the extension increments are quite small. The raw data in most cases contains apparent jumps in crack length, a phenomenon which is believed to result from the difficulty of optically resolving small extensions and/or crack tunnelling in the specimen bulk underneath the surface cladding.

Established methods, such as optical and DC potential drop, for measuring crack length in fatigue and fracture experiments are suitable for measuring a single, identified crack of

substantial length. However, these methods do not give accurate results when applied to much smaller cracks and potential crack sites in an MSD situation.

Alternating current potential drop (ACPD) equipment and procedures have been evaluated for use in measuring small cracks typical of MSD [23]. Controlled laboratory experiments have been conducted in which the ACPD method was used to measure crack length. The test specimens were subsequently fractured, and the actual crack lengths were verified fractographically. The tests indicate that the ACPD technique can be used to measure crack growth, but that difficulties in calibrating the instrumentation may limit its practical application.

### Material Behavior Characterization

A better understanding of material behavior in the presence of MSD will improve the accuracy of analysis. Both conventional and advanced fractographic techniques have been used to characterize the material behavior that can have an effect on MSD.

The fracture features of riveted lap joint specimens tested in fatigue have been analyzed fractographically [24]. The investigation included a count of fatigue striation, spacing and density, assessment of the fracture mode in the cladding near the surface, and review of the mode of failure (plane stress versus plane strain). The crack growth rates were determined from the striation spacing and reported as a function of maximum stress values. The plastic zone size was correlated with the stress intensity factor calculated for the assembly being tested.

MSD crack growth has also been investigated using an advanced quantitative fractographic technique known as FRASTA (Fracture Reconstruction Applying Surface Topographic Analysis) [25]. A three-hole specimen with crack-starter notches at the center hole was tested, first under fatigue cycling, then by applying a monotonically increasing load until the specimen failed. Analysis of the fracture surfaces produced crack profiles and fractured area projection plots for both fatigue and stable crack growth. The results revealed significant tunnelling of the crack front during the early stages of fatigue cycling, and that the crack grew at a constant opening angle during stable crack growth under monotonic loading. A finite element analysis was used to confirm the fractographic results, and to calculate the stress intensity factor and the J-Integral.

The FRASTA technique will be utilized next to compare the cracked panels taken from airplanes with those from laboratory tests of both recently made and older 2024-T3 aluminum panels. It will be applied to the fracture surfaces of fatigued panels to reconstruct the fracture events, determine microfailure mechanisms, and crack tip characteristics such as the



displacement and angle of the crack tip opening. The results are expected to indicate whether differences exist in the failure behavior of aircraft panels and panels tested under cyclic loads in the laboratory. Thus, the findings will help justify the use of laboratory test data for designing aircraft structures, setting inspection schedules, and defining maintenance procedures and may lead to improved laboratory tests that more reliably represent service conditions.

### Fatigue Crack Growth in 2024-T3 Aluminum

Future work on material behavior characterization will be conducted through fatigue tests using standard specimens with 0.04-inch thickness. Particularly, the effects of stress ratio (the ratio of minimum to maximum stress), cycle frequency, strain rate, and environment will be individually explored in several series of tests.

The effect of stress ratio on fatigue crack propagation will be examined through tests performed in air. The cycling frequency will be held constant at 10 cycles per second (cps) while the stress ratio is varied. Four values will be examined: 0.05, 0.3, 0.6, and 0.8.

The effect of cycling frequency will be investigated by comparing macroscopic crack growth rates and fracture surface striation (beach marking of the fracture surface) for tests at 0.01, 10, and 20 cps. Some preliminary results indicate that the effect of frequency is negligible.

Additional tests will be conducted to determine strain rate effects on 2024-T3 alclad aluminum tensile flow stress. As part of the cycle frequency effect investigation, monotonic stress-strain relationships will be determined for strain rates varying by several orders of magnitude (0.001, 0.04, and 0.34 percent per second).

The contribution of environment to fatigue crack propagation will be evaluated by submerging crack growth specimens into aerated 3% sodium chloride (NaCl). The tests will be performed at relatively high (10 cps) and moderate (0.1 cps) frequencies. Clad and clad-removed specimens will be exposed to aerated 3% NaCl for varying lengths of time and anodic polarization voltages to establish corrosion behavior and accelerating pitted behavior.

### INSPECTION FOR MSD

Proper maintenance and inspection are keys for insuring the safety of aging airframes with MSD potential. Preliminary tests and calculations suggest that MSD must be detected at quite small crack lengths and in much shorter time than an isolated crack, if MSD is to be found and repaired ahead of linkup and fracture.



The issue of inspection involves both inspection techniques and inspection interval.

### Alternative Measurement Techniques

The requirements of detecting small cracks for MSD preclude reliance on visual inspection only. The only alternative which is in common usage in the airline industry involves the use of hand held eddy current probes. The eddy current method is technically reliable but tedious to apply, leading to excessive down time and human factors problems. Better nondestructive inspection (NDI) methods must be sought to provide airlines with procedures which are both appropriate for MSD and economical to apply.

New NDI technologies under consideration include infrared imaging and shearography, which have the potential to inspect a large area at a time. The capability to perform large-area inspection is especially desirable when considering the large number of rivets in a typical lap joint. Also, techniques for detecting disbonds in a lap joint are being investigated.

Under a Cooperative Research and Development Agreement between VNTSC and Henson Aviation, Inc, operator of US Air Express, a shearographic demonstration inspection (Figure 19) of some portions of a Boeing 737 aircraft fuselage was performed at a US Air repair station at Winston-Salem, NC, during August 21-23 1991. The inspection concentrated on comparing effectiveness of shearography with currently mandated methods in detecting disbonds in the fuselage.

Adhesive bonding is utilized in modern aircraft fuselages, frequently in combination with rivets. As aircraft age, bond failure becomes a major problem, since it may promote fatigue cracking, moisture intrusion, and subsequent corrosion. Any of these events may cause cabin pressure loss and, possibly, catastrophic fuselage failure.

The shearographic method of detecting disbonds depends on the deformation of the aircraft skin under varying pressurization. When illuminated by coherent light, the phase relationship and intensity of the light reflected from any two points of the skin changes as a result of this deformation. Figure 19 shows the instrument being used to inspect a lap joint for disbonds. Surface changes down to 0.00025 millimeter can be detected and displayed as a real-time image of the field of view. Comparison of successive images as the pressure changes permits interpretation of the condition of a bond.

For the portions of specific interest of the fuselage examined in this demonstration, 31 disbonds were found by shearography; 25 were confirmed by ultrasonic inspection. Of the remainder, five were disbonds on repaired riveted lap joints

where the ultrasonic device cannot perform reliably, and one was a disbond on a riveted stringer which the ultrasonic device did not detect for the same reason. In addition, there was one ultrasonic device false positive confirmed by reference to a drawing, and by observation.

The demonstration indicated potential advantages of shearography over currently used inspection techniques, specifically, potential for improved reliability in the detection of disbonds in the fuselage and reduced down-time of the aircraft while reducing inspection costs.

In a parallel activity, a library has been established which contains a variety of specimens, including pieces taken from actual airplanes as well as panels that have been used in testing. Some specimens were selected as examples of design features found in airplanes, while others display the major types of damage encountered in the aging airplane fleet, including cracks, corrosion, and disbonding. The specimens are used in calibrating NDI instruments, evaluating new techniques, and training NDI personnel.

#### Proof Testing

In mid-1988 a number of the independent experts who advise the FAA on structural integrity suggested that proof testing be considered as an interim backup option, to be implemented if the NDI program should prove to be ineffective, until the terminating actions specified by the Airworthiness Directives (ADs) have been applied to those airframes. The proof test is an appealing concept because it appears to eliminate the uncertainties of NDI by establishing a precise upper limit on existing crack size, relative to the critical size in flight. The proposal recommended a pressure test to limit load, i.e. 1.33 times the normal inflight pressure differential (1.33P).

The FAA/VNTSC and the National Aeronautics and Space Administration (NASA) have carried out an independent technical evaluation of the concept of pressure proof testing [26-28]. The objectives of the evaluations were to establish the potential benefit of the pressure proof test, to quantify the most desirable proof test pressure, and to quantify the required proof test interval. The focus of the evaluations was on multiple-site cracks extending from adjacent rivet holes of a typical fuselage longitudinal lap splice joint.

The FAA/VNTSC evaluation [26] involved a damage tolerance analysis of the 737 fuselage structure because of the availability of MSD crack growth data pertinent to the type of situation for which the proof test had been proposed. The effects of stress, proof pressure load, material data, rivet hole size and rivet spacing were assessed. Preliminary R-curve properties derived from the FAA/VNTSC laboratory test program



data were used to estimate the critical crack lengths corresponding to proof and maximum service pressures. A range of proof pressure was studied, not only for the effect on post-test safe crack growth interval, but also to investigate the potential for stable crack extension during the test itself. One experimental evaluation conducted under the FAA/VNTSC research program simulated the 1.33 proof factor and verified the inspection interval of the analysis.

NASA's investigation also involved a combined experimental and analytical study. Tests were conducted on panels with a long central through-crack to simulate MSD after linkup. Tests were also conducted on panels with evenly spaced unloaded holes and panels with a lap splice joint attached by a single row of rivets to simulate MSD before linkup.

The results from the two independent evaluations are summarized in Table 5 showing the required proof test interval for proof factors of 1.33 and 1.50. The proof factor is defined as the ratio of the proof test pressure load divided by the normal in-flight pressure load. The conventional factor-of-safety of 2.0 has been applied to the proof test intervals to achieve the results in Table 5. The safety factor compensates for the uncertainties involved in making crack growth life predictions.

A summary of the more general qualitative results obtained from both investigations [28] is given below.

1. The remaining life with the proof test is longer than without the proof test for a proof factor of 1.33 to 1.5.
2. The remaining life after the proof test increases with increasing proof factor up to a value about 1.5.
3. The FAA evaluation revealed that equal safety to that of proof testing could be achieved by eddy current inspection of the rivets in the splice joints at an inspection interval of about 1200 flights.

The following conclusions can be drawn from the two studies.

1. For a proof factor of 1.33, the required proof test interval must be below 300 flights to account for uncertainties in the evaluation.
2. For a proof interval of 300 flights, the proof test must be repeated on a regular basis within the period of the terminating action.
3. Conducting the proof test at a proof factor of 1.5 would considerably exceed the fuselage design limit



load and, therefore, is not consistent with accepted safe practice.

4. Better safety can be assured by implementing enhanced NDE inspection requirements and adequate reliability can be achieved by an inspection interval several times longer than the proof test interval.

As a result, pressure proof testing of the fuselage of aging commercial transport aircraft is not recommended.

### Inspection Interval [7]

One key element of a successful inspection program is the interval between inspections. Too short an interval becomes economically burdensome, while too long an interval increases the possibility that a critical crack will go undetected. The selection of inspection interval (or strategy) should, therefore, be based on the required level of safety.

The cumulative probability of detection of critical cracks determines the level of safety of an inspection program. However, the decision about the required cumulative probability of detection is a difficult one. The problem becomes more apparent when it is phrased in terms of the probability that the crack will be missed. Requiring, for example, that the inspection interval be selected to provide a cumulative probability of detection of 0.98 before the crack reaches  $a_p$ , maximum permissible crack, means that it is acceptable that the crack is missed in 2% of the cases.

The conventional selection of inspection interval does not explicitly define the probability of missing a critical crack. In the conventional approach, the damage tolerance analysis is used to determine the safe crack growth period,  $H$ , which is defined as the time required for a crack subjected to a given stress spectrum to grow from the detectable to  $a_c$ , the critical size, or  $a_p$ , the permissible size. Then  $I=H/2$  is taken as the inspection interval. This approach provides for two inspections during which a potential critical crack can be discovered and repaired. Depending on the growth characteristics and the nature of the crack, the probability of detecting the crack is different.

The above can best be illustrated by an example. Consider Figure 20, showing two crack growth curves. Assume both cracks have the same maximum permissible size  $a_p$  and  $H$ . Both will be inspected twice between  $a_d$ , detectable size, and  $a_p$ . At both inspections the case 1 crack is larger than the case 2 crack, so that the second has a higher probability of being missed than the first. For this example the crack growing fastest initially is safer by having a higher chance of being discovered.

The prediction of MSD crack growth and detection is needed in order to relate the cumulative probability of detection to different inspection intervals. A numerical model has been developed based on conventional nondestructive inspection techniques [29]. Three simple, but common, configurations were considered for illustration: (1) cracks emanating from a rivet hole; (2) crack emanating from a (large) structural hole; and (3) a corner crack in a heavy member. Crack growth was calculated using the TWIST standard spectrum [30], concurrent with the above limit load stress. Taking a case of easy access and high specificity of inspection, and taking the probability of detection (POD) curves following from a re-assessment [31-33] of the data [30], the cumulative probability of detection was calculated as a function of the length of the inspection interval, assuming visual inspection for the first two cases, and ultrasonic inspection for the third. The results are shown in Table 6.

It should be noted that this is an example only. To prevent unwarranted conclusions, the real configurations are not identified in Table 6. The cases considered were merely labelled A, B, and C, while the order is different from 1, 2, and 3 above. Table 6 shows the required inspection intervals for different cumulative probabilities of detection.

It is proposed that work be performed to obtain in-situ measurements of the POD curves for all inspection procedures used, with due account for accessibility and specificity, and to determine the cumulative probability of detection (CPOD) of critical cracks. Such information is essential to determine the associated risk of non-detection and to establish the relationship of the damage tolerance requirements (DTRs) to the safety of aging aircraft.

It is important to re-examine these DTRs so that the problems of MSD associated with aging aircraft can be avoided for the future generation of aircraft. One basis for a review of the DTRs could be a risk analysis to assess the probability of structural failures and to determine the inspection interval. This should lead to either one of the following improvements [7]:

- a. Provide a definition of detectable crack size in terms of POD, if the present practice of  $I = H/j$  is to be continued, and an associated definition of the safety factor  $j$ .
- b. Instead of (a), require that the inspection interval be determined for a certain fixed cumulative probability of detection. This eliminates the need for a definition of the detectable crack size and for specification of a safety factor.

We shall briefly describe the risk analysis method and its required data items in the following section.



## Risk Analysis

The risk analysis uses a probabilistic approach to determine the probability of failure of the single flight of a single airplane at a given time. A risk analysis [7,34] requires the following data items at the location of interest in an aircraft structure:

- crack size probability distribution,
- stress probability distribution,
- critical crack size vs. stress,
- crack growth curve, and
- inspection probability of detection (POD) curve.

The POD depends heavily on the inspection method, the accessibility of the location, and the specificity of the inspection. The crack size distribution is also a function of the age of the aircraft. These data items are needed for all critical locations of the airframe in order to perform the risk analysis for the aircraft.

The crack populations at critical locations in the structure are usually difficult to obtain. Even when the data are available, they are only the data at some point in time of the service life of the aircraft. A crack growth analysis must be performed to derive the crack lengths as a function of flight hours. The crack distributions are generally obtained from destructive teardown and/or detailed inspections of the critical components.

The second required data item is the probability of exceeding a given stress at all critical locations in a single flight. This information is needed to calculate the probability of exceeding the residual strength in the flight (i.e., the probability of failure) and to predict the crack size distributions for later flights.

Crack growth prediction is normally based on fracture mechanics and is sensitive to loads (stresses and stress sequence). One way to improve the fracture analysis is to obtain more information on past loading by instrumenting aircraft to record load or stress history. The better one knows the past, the better one can anticipate the future. With the introduction of smaller and more sophisticated chips, it will become possible to equip each aircraft with a load/stress recorder. This will provide a better basis for analysis because effects such as clipping can change crack growth by factors of two and three.

The third required data item, critical crack size versus stress, is related to the residual strength of the aircraft. This information and the curve of crack growth versus flights will have to be determined by tests, a fracture analysis, or a combination of the two.



By integrating the probability of stress levels over the domain where the crack size exceeds the critical crack size for that stress, we obtain the probability of failure for the location. If the probabilities of failure of different locations are statistically independent, then the single flight probability of failure is one minus the product of the probability of no failure at all locations.

The risk assessment methodology is useful for analyzing aircraft components nearing the end of their useful lives. The single flight probability of failure provides us with an instantaneous information of the risk at some point of the aircraft's life. This interpretation of the quantity may be difficult and requires further work. Lincoln [34] suggested to relate the quantity to the risk we accept in our everyday living. For example, the risk of a major accident that we accept in driving an automobile to work and back home is of the order of  $10^{-6}$ . Another of his suggestions is to interpret the quantity based on the precedents that have been set for other aging aircraft. He stated that, for most military systems, a single flight probability of failure of  $10^{-5}$  or greater for an extended period of time is considered unacceptable. Whether or not these numbers are suitable for commercial transport category airplanes needs to be evaluated.

The probability of failure after a given number of flight hours should be information very valuable to airline operators and the FAA. This information is also useful for judging the effectiveness of an inspection strategy and the effect of changes in an inspection program. As of now the data required for the risk analysis of commercial transport category airplanes are not complete. The task of carrying out a rigorous risk analysis for all the airplane models is a tremendous undertaking and yet to begin.

#### MSD AVOIDANCE

The FAA research program not only has responded to the problems associated with MSD potential in existing fleets, but also has the objective of fostering improvement of practices to avoid MSD in the future. This goal applies to repairs as well as new designs.

A conceptual model has been developed to rank MSD potential of alternative design details [35]. The model combines fatigue and damage tolerance analyses in a practical engineering tool which yields a risk parameter. While the model is conceptually straightforward, its application requires extensive testing of realistic details subjected to realistic stress environments to provide for appropriate estimates of the fatigue life parameters.

## Repair Practices

Repairs and major structural modifications often can change the MSD potential of an airframe. The general practice is to have a repair to possess static strength equal to that of the original airframe. Such a repair can become a site for later fatigue damage in the repaired or an adjacent area unless damage tolerance is also considered in the repair design.

Airframe structures which are damaged are often repaired by riveting a doubler over the damaged area. The rivets provide a load path mechanism for transfer of load from the skin to the doubler. The bearing stress induced at the rivet holes, as well as the stress concentration on the holes, will degrade the fatigue life of the skin. Reduced inspectability of these rivet holes due to doubler coverage may also be a concern. Therefore, to design a damage tolerant repair, it is essential that the calculation of loads and stresses be done to sufficient detail and accuracy so that the most critical fatigue locations can be determined and the fatigue life evaluated. The hybrid finite element method is well suited to solving repair problems. The hybrid super element accurately models the stiffened cracked panel with rivet holes, while the standard finite element method provides the versatility to take into account a variety of doubler designs, such as single doublers, two-sided doublers, laminated doublers, and finger doubler configurations [19]. An alternative approach to the analysis of riveted doubler repairs is the displacement compatibility method of Swift [36], and the recent extension of this method to two-dimensions by Battelle (under contract to VNTSC) with the program "SKINFIX". Based on the results in references [19] and [36] it can be shown that the highest rivet loads occur at the first attachment row of the doubler, with the maximum load at the corner rivet, along with the maximum skin bearing stress. Skin fatigue life can be estimated from open hole S-N data, suitably modified for stress effects using the ratio of bearing stress to gross stress. Parametric studies show the effect of doubler thickness. In general, increasing doubler thickness decreases fatigue life. Reductions in fastener load can be achieved by using multiple doublers (lamination), whether the secondary doubler is on the outside or the inside of the skin. Another technique for reducing rivet loads and consequently increasing skin fatigue life is through the use of a finger doubler configuration.

The repair of cracked structures with a bonded composite patch is a promising concept. A unidirectional composite bonded across a crack can significantly limit the crack opening and thus the stress intensity factor and crack growth. Any patch will reduce the net stress in the damaged region, but a bonded patch does not increase the number of stress concentration points (i.e., holes) as a patch fastened with rivets would. A highly orthotropic composite will not profoundly increase the transverse stiffness and thus will not significantly affect the transverse load paths. Some composite patch materials such as boron/epoxy



can provide protection against corrosion while not disrupting some nondestructive inspection (NDI) methods such as eddy current monitoring.

Analytical tools are being developed to quantify some of the benefits of composite patches [37]. Analyses show the reduction in stress intensity at the crack tip as a result of patch parameters. For example, the patch geometry shown in Figure 21 leads to the reductions in stress intensity factors shown in Figure 22. Crack geometries examined include center cracks, edge cracks, surface cracks, through cracks at the edge of loaded holes, and surface cracks at the edge of loaded holes. The lower stress intensity factors imply slower crack growth rates in these cases. This inhibited growth has been verified experimentally for some of these geometries, with specimen lives increasing by factors of up to 22. A patch has also been designed for a simulated typical lap joint with multiple site damage and has shown life improvements. This implies that composite patches may be a viable solution.

#### CONCLUDING REMARKS

The research progress to date and future plans are summarized as follows:

##### Understanding MSD Behavior

We have proposed a three-part definition to more precisely define MSD. We have also constructed a special fixture and tested the residual strength and fatigue life of cracked full scale curved panels 68 inches on the circumference by 120 inches axially. Additional tests are being conducted to demonstrate the residual strength of curved panels with and without MSD cracks in the lap splice. Also, additional fatigue tests will be conducted to determine multiple site crack growth behavior and the effectiveness of the terminating action on curved fuselage panels.

Preliminary tests indicated that tear straps were effective in providing fail-safety for running midbay cracks without MSD by causing flapping to occur, thereby containing or arresting the cracks. However, the crack did not turn at the tear straps in the two tests involving lap splice cracks, one with and the other without MSD. That is, the lap splice cracks extended beyond the tear strap in the same direction as the running crack. In parallel with the curved panel test effort, analytical models based on the compatible displacement method have been developed to predict the failure pressure of the tests. The method can readily account for the effects of rivet flexibility, biaxially applied stress, stiffener bending, and linear and nonlinear rivet flexibility (as measured in separate experiments). Good



comparisons between analytical and experimental results have been obtained.

A fatigue test indicated that cracks could be formed on rivet holes at locations other than the upper row of rivets on the outer skin. This could be due to effects of excessive local bending at those locations. Further investigations will be made to resolve the issue. The test also showed that cracking in the tear straps occurred at the same time MSD was developing which may further degrade the fail-safe capability of the structure.

We have successfully reproduced MSD in tests using flat 12-inch coupons at a life reasonably consistent with what has been observed in service. The 12-inch panel tests also demonstrated that the terminating action mandated by the FAA is effective in reducing the growth rate of cracks in the upper row of rivet holes. Full-scale curved panel tests will be conducted to confirm these tests. The test results also showed that an added shear load could cause a reduction in fatigue life and that stress level, rivet type, and skin thickness had a significant influence on fatigue. Future experimental work will be conducted on lap joint specimens to determine the effect of pre-existing corrosion on fatigue initiation and crack growth behavior.

Improved analysis methods to predict the tolerance of airframe structures to MSD are being developed. Preliminary results indicate that both the hybrid finite element method and the alternating methods are efficient and effective in accounting for the effects of complex geometry variations and different fastener configurations. These methods can also account for stress singularities at crack tips and rapid stress variations near rivet holes. Methods to account for the effects of yielding, curvature, and bulging on MSD are currently being investigated.

Test data suggests that the R-curve for simulated MSD depends on initial crack length as well as extension, i.e., there was apparently no "master" curve independent of initial crack length, as can be found in conventional R-curve tests. The strain energy density criterion was used in conjunction with elastic-plastic finite element analysis to predict the stable crack growth observed in panels with simulated MSD [22]. The results show that the strain energy density analysis is a viable approach to damage tolerance analysis of MSD where the stability limit is controlled by plastic collapse or ductile fracture.

The fracture features of riveted lap joint specimens tested in fatigue have been analyzed fractographically [24]. The crack growth rates were determined from striation spacing and reported as a function of maximum stress values. MSD crack growth has also been investigated using an advanced quantitative fractographic technique known as FRASTA [25]. Analysis of the fracture surfaces revealed significant tunneling of the crack front during the early stages of fatigue cycling, and that the

crack grew at a constant opening angle during stable crack growth under monotonic loading. The results were confirmed by a finite element analysis. The analysis was also used to determine the stress intensity factor and the J-Integral. The FRASTA technique will be used next to compare the cracked panels from the field with both recently made and older 2024-T3 aluminum panels.

### Inspection for MSD

The requirements of detecting small cracks for MSD preclude reliance on visual inspection. Alternative measurement techniques are being assessed. New NDI technologies under consideration include infrared imaging and shearography, which have the potential to inspect a large area at a time. A demonstration of shearography as a method to inspect for disbonds was performed on the fuselage of a Boeing 737. The demonstration indicated potential advantages of shearography over currently used inspection techniques, such as improved reliability in the detection of disbonds in the fuselage, reduced downtime of the aircraft, and reduced inspection costs.

A library has been established which contains a variety of specimens of different design features with the major types of damage encountered in the aging airplane fleet, including cracks, corrosion, and disbonding. The specimens are used in calibrating NDI instruments, evaluating new techniques, and training NDI personnel.

The evaluation of the concept of pressure proof testing has been completed. The results indicated the required proof test interval is too short to be practical.

A methodology based on the risk analysis has been presented showing the procedures for establishing the inspection requirements. It is proposed that work be performed to obtain in-situ measurements of the POD curves for all inspection procedures and to determine the cumulative probability of detection of critical cracks. Such information is essential to determine the associated risk of non-detection and to establish the relationship of the damage tolerance requirements to the safety of aging aircraft.

### MSD Avoidance

A conceptual model has been developed to rank the MSD potential of alternative design details [35]. While the model is conceptually straightforward, it requires extensive fatigue data from samples replicating realistic structural details and loading conditions.

Repairs and major structural modifications often can change the MSD potential of an airframe. Riveted doubler repairs can



degrade the fatigue life of fuselage skin unless proper care in design detail is taken. Design of repairs to an equal or better static strength capability is not sufficient for predicting fatigue life; damage tolerance principles must be used and evaluated. Reduced inspectability due to doubler coverage is also a concern. In order to fully evaluate a repair, analytic tools which are accurate and efficient must be used. VNTSC has used three analysis techniques: hybrid finite element, one-dimensional displacement compatibility, and the two-dimensional extension. These programs have been evaluated and results compared in order to fully assess all aspects of riveted repairs.

The repair of cracked structures with a bonded composite patch is being evaluated. A composite patch bonded across a crack can significantly reduce the net stress in the damaged region. This inhibits crack growth which has been verified experimentally for some geometries, with specimen lives increasing by factors of up to 22. Some composite patch materials such as boron/epoxy can provide protection against corrosion while not disrupting some nondestructive inspection (NDI) methods such as eddy current monitoring.

The FAA research program on structural integrity has gone a long way toward understanding the behavior of and controlling MSD. Many technical issues are being actively pursued. We will continuously report the progress as the research results are available.

#### ACKNOWLEDGEMENTS

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TABLE 1. TEST CONFIGURATIONS FOR WORK RELATED TO CAUSES  
AND LIKELIHOOD OF MSD

Config- uration No.	Stress (ksi)	Rivet Type	Rivet Spacing (in.)	Rivet Orien- tation <sup>1</sup>	Number of Rows	Skin Thick- ness (in.)
1	12	Flush	1.00	S	5	0.040
2	12	Briles	1.00	S	3	0.050
3	12	Flush	1.00	C	5	0.063
4	12	Briles	1.29	C	5	0.040
5	12	Flush	1.29	S	5	0.050
6	12	Flush	1.29	S	3	0.063
7	12	Flush	0.75	S	3	0.040
8	12	Flush	0.75	C	5	0.050
9	12	Briles	0.75	S	5	0.063
10	14	Briles	1.00	C	3	0.040
11	14	Flush	1.00	S	5	0.050
12	14	Flush	1.00	S	5	0.063
13	14	Flush	1.29	S	5	0.040
14	14	Flush	1.29	C	3	0.050
15	14	Briles	1.29	S	5	0.063
16	14	Flush	0.75	S	5	0.040
17	14	Briles	0.75	S	5	0.050
18	14	Flush	0.75	C	3	0.063
19	16	Flush	1.00	S	5	0.040
20	16	Flush	1.00	C	5	0.050
21	16	Briles	1.00	S	3	0.063
22	16	Flush	1.29	S	3	0.040
23	16	Briles	1.29	S	5	0.050
24	16	Flush	1.29	C	5	0.063
25	16	Briles	0.75	C	5	0.040
26	16	Flush	0.75	S	3	0.050
27	16	Flush	0.75	S	5	0.063

1 Rivet Orientation: S = Staggered, C = Continuous

TABLE 2. RESULTS OF STATISTICAL SIGNIFICANCE TEST

Parameter	F Level of Significance
Stress	0.0056
Rivet type	0.0069
Skin thickness	0.0179
Rivet spacing	0.1468
Rivet orientation	0.1489
Number of rows	0.1544
Interaction of stress with rivet spacing	0.3050
Interaction of Stress with skin thickness	0.7640
Interaction of rivet spacing with skin thickness	0.9437

TABLE 3. STRESS INTENSITY CORRECTION FACTOR  $\beta$  for STIFFENED PANEL WITH SINGLE CRACK ( $K=\beta\sigma_{II}^0\sqrt{\pi a}$ )

<u>a(in.)</u>	Hybrid Element (present)	Hybrid Element (Tong'84)	Compatible Displacement (Swift)
.375	1.58	1.58	1.56
1.125	1.31	1.31	1.38
3.000	1.17	1.17	1.21
6.000	1.11	1.11	1.12
9.000	1.06	1.06	1.06
12.000	0.848	0.849	0.869
13.500	0.699	0.682	0.713
15.000	0.710	0.681	0.696
16.500	0.705	0.688	0.695

Table 4. STRESS INTENSITY FACTORS FOR STIFFENED  
 PANEL WITH TWO CRACKS  
 (per unit far field stress)

$K_I(ksi\sqrt{in})$			
<u>a(in.)</u>	<u>2a/d</u>	<u>Outer tip</u>	<u>Inner tip</u>
0.3	0.05	1.6110	1.6121
0.6	0.1	2.0018	2.0051
1.2	0.2	2.6136	2.6249
1.8	0.3	3.0622	3.0908
2.4	0.4	3.4355	3.4960
3.0	0.5	3.8019	3.9194
3.6	0.6	4.1551	4.3741
4.2	0.7	4.4977	4.9034
4.8	0.8	4.8352	5.6172
5.4	0.9	5.2538	7.0135
5.7	0.95	5.5581	8.7794
5.9	0.98	5.8346	12.6465

d = distance between crack centers = 12-in.



TABLE 5. REQUIRED PROOF TEST INTERVAL TO SCREEN CRITICAL  
MULTIPLE-SITE CRACKING IN RIVETED SPLICE JOINTS

Evaluation	<u>PROOF FACTOR</u>	
	1.33 <u>(# of Flights)</u>	1.50 <u>(# of Flights)</u>
NASA	275	765
FAA	200	600

TABLE 6. INSPECTION INTERVALS AND CUMULATIVE PROBABILITY OF  
DETECTION FOR THREE CONFIGURATIONS [7]

Confi- guration	Inspection interval (hrs) for cumulative probability			I=H/2	Cumulative probability
	<u>0.95</u>	<u>0.98</u>	<u>0.99</u>	<u>I (hrs)</u>	<u>for H/2</u>
A	3,000	2,250	2,100	3,750	0.90
B	1,250	1,000	800	1,250	0.95
C	5,700	4,500	4,000	7,500	0.89

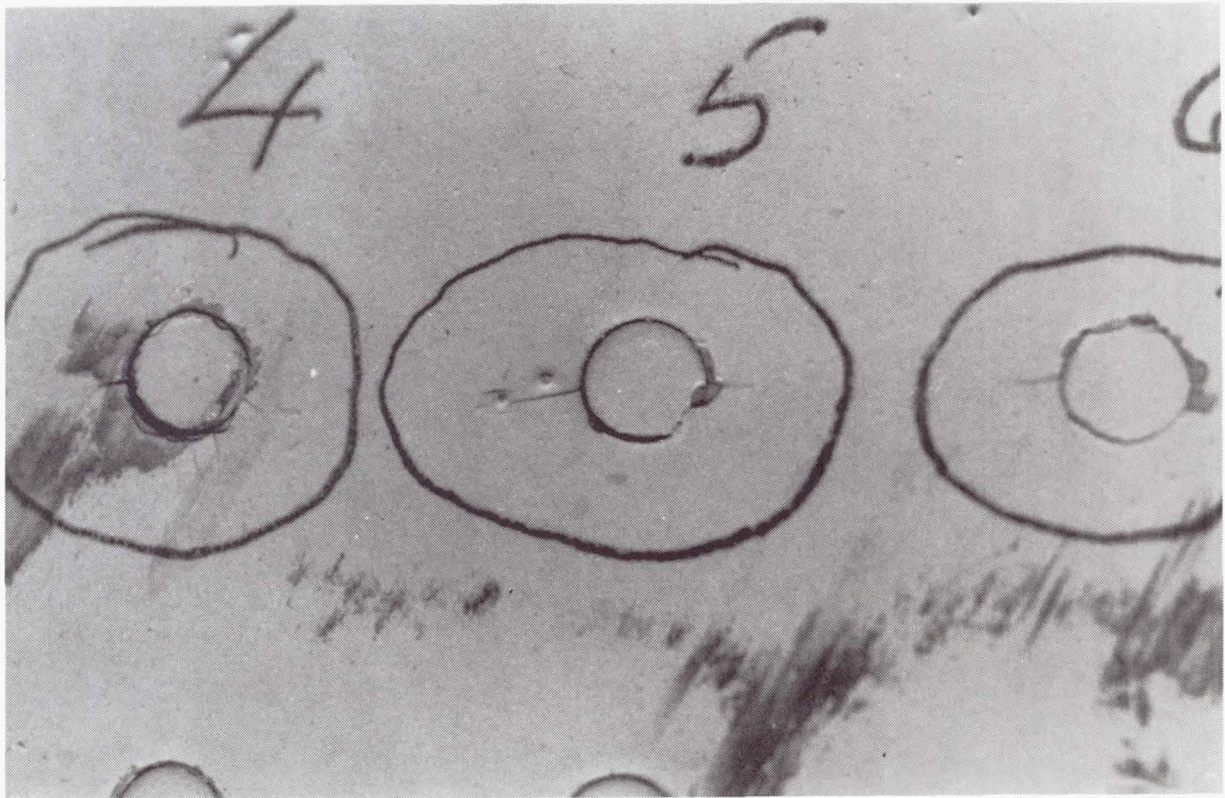


FIGURE 1. MULTIPLE SITE DAMAGE IN N73711  
(FROM REFERENCE [3], COURTESY OF T. SWIFT).



FIGURE 2. SCHEMATIC OF A TYPICAL MSD CONFIGURATION.

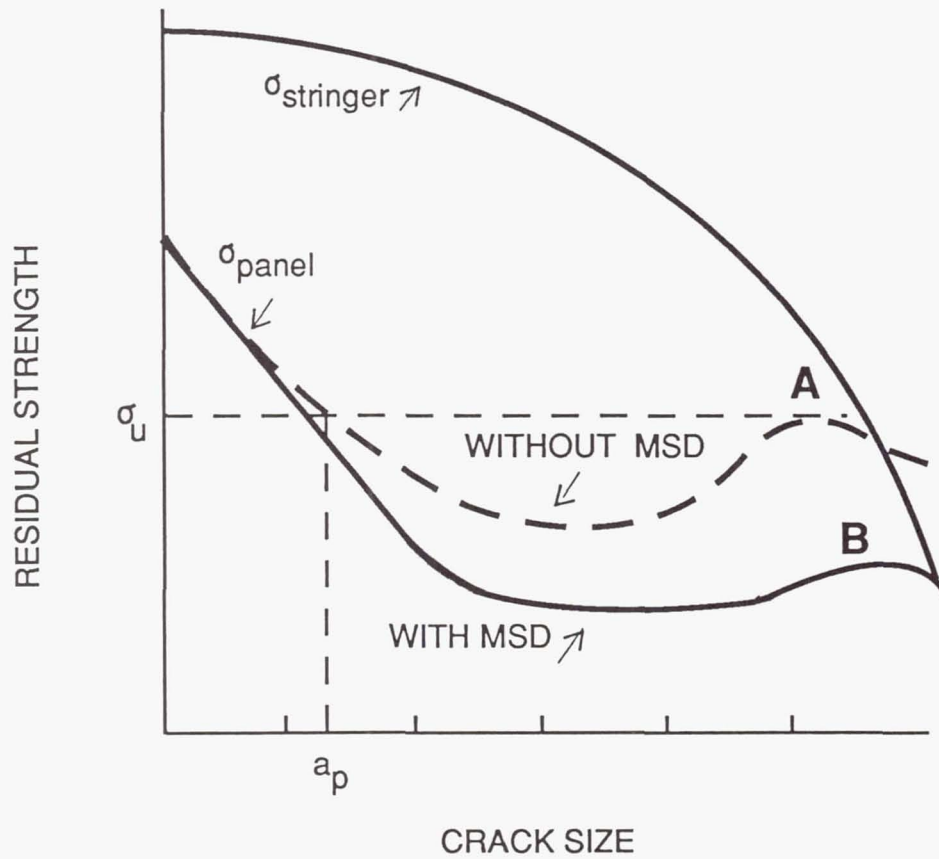
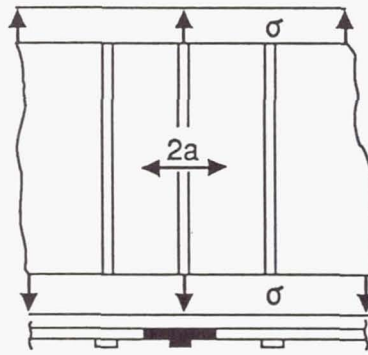


FIGURE 3. EFFECT OF MSD ON RESIDUAL STRENGTH  
(REFERENCE [7])



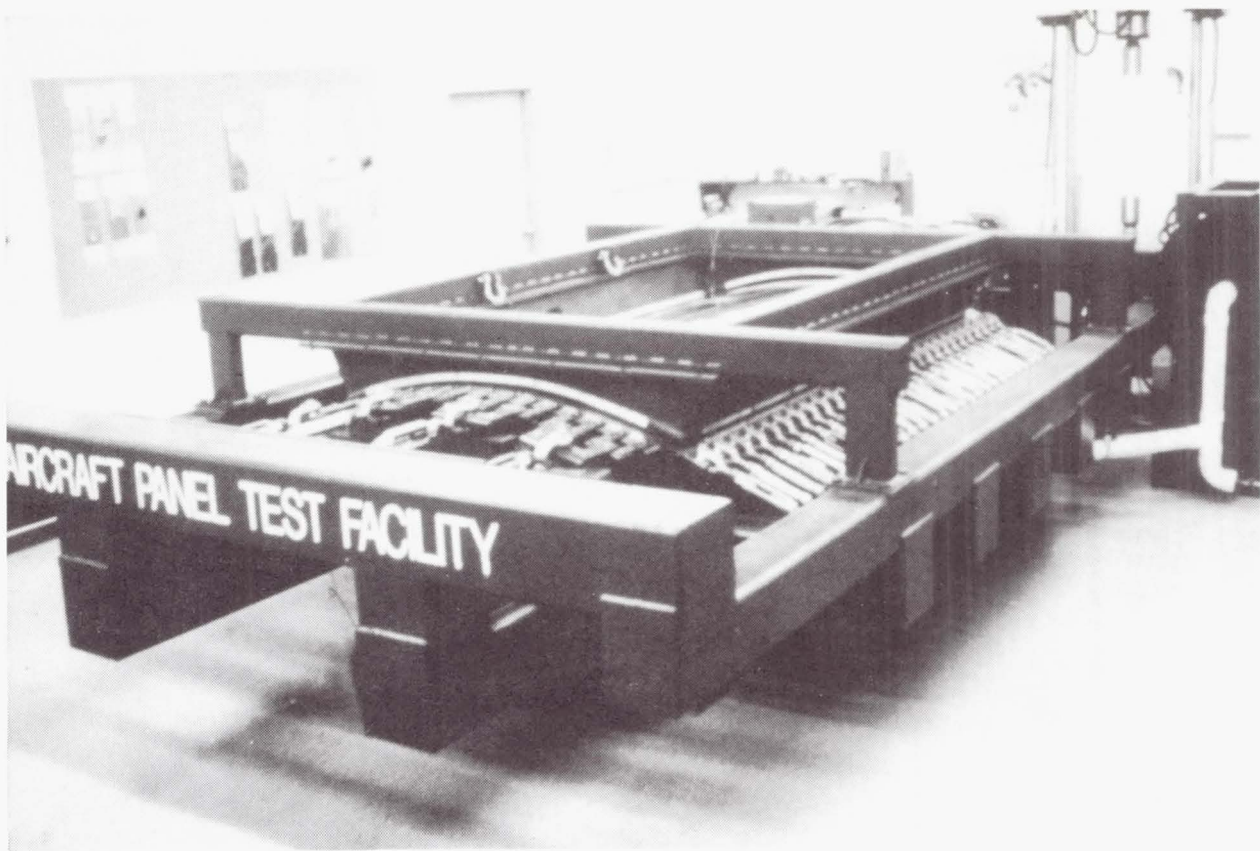
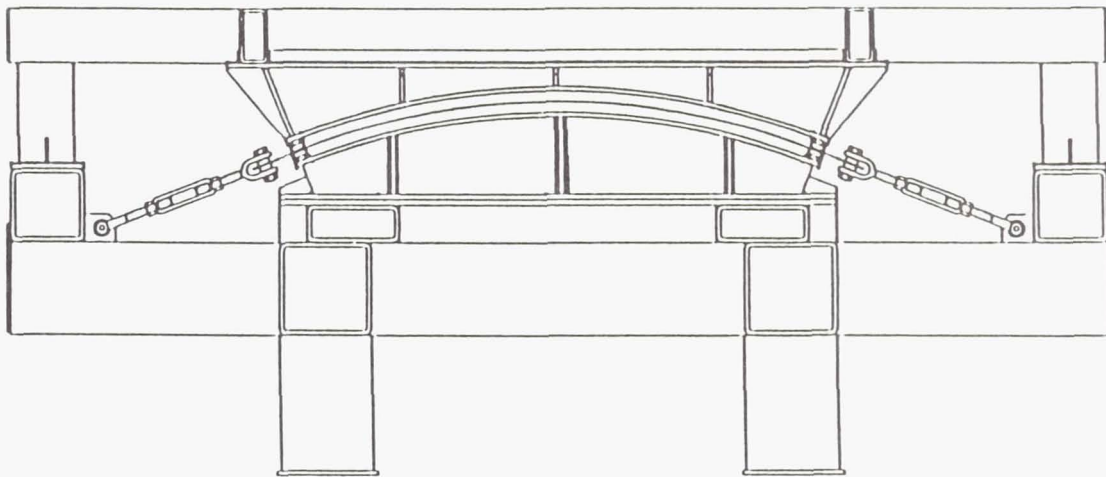


FIGURE 4. CURVED PANEL TEST FIXTURE

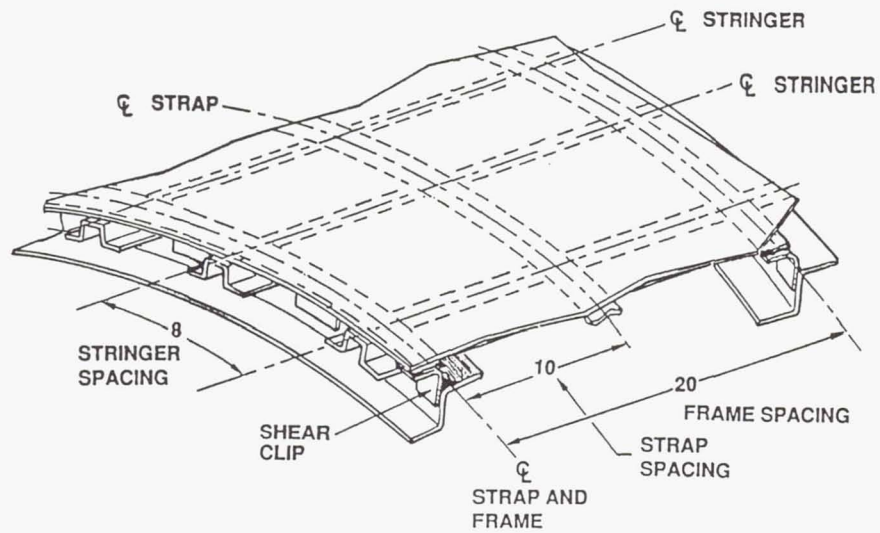


FIGURE 5. STRUCTUAL DETAILS OF CURVED TEST PANEL

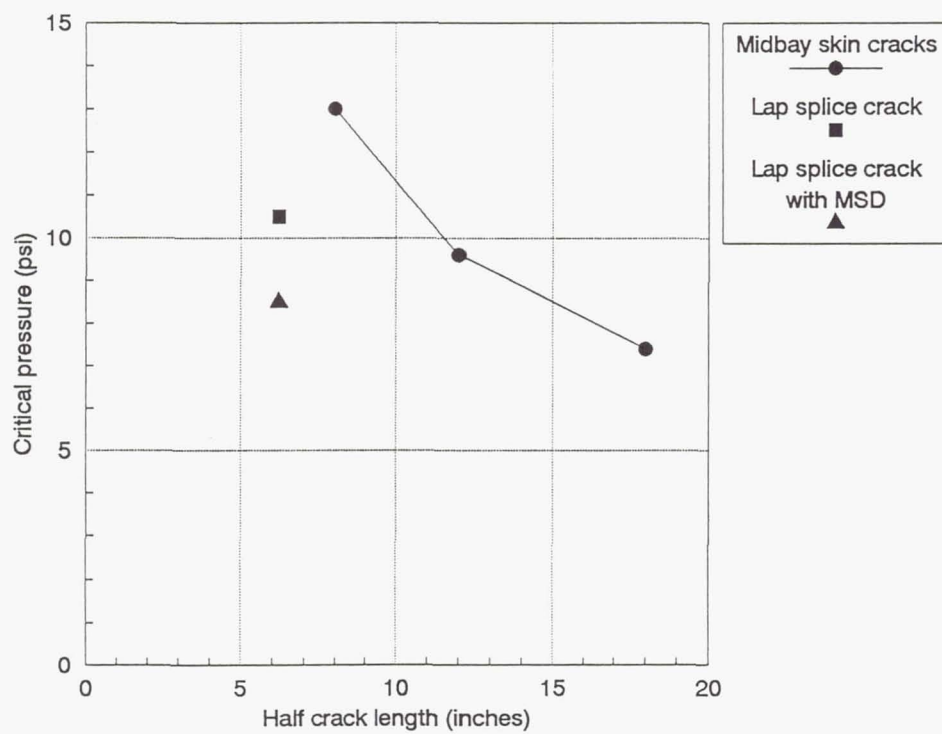


FIGURE 6. CURVED PANEL SHAKEDOWN TEST RESULTS

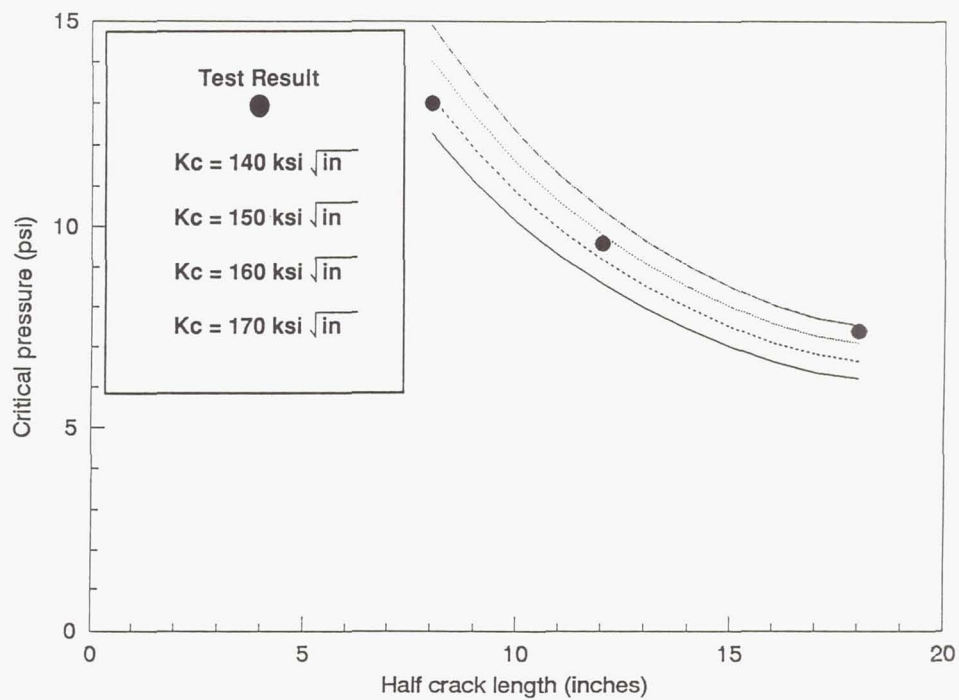


FIGURE 7. COMPARISON OF CURVED PANEL TESTS AND ANALYSIS

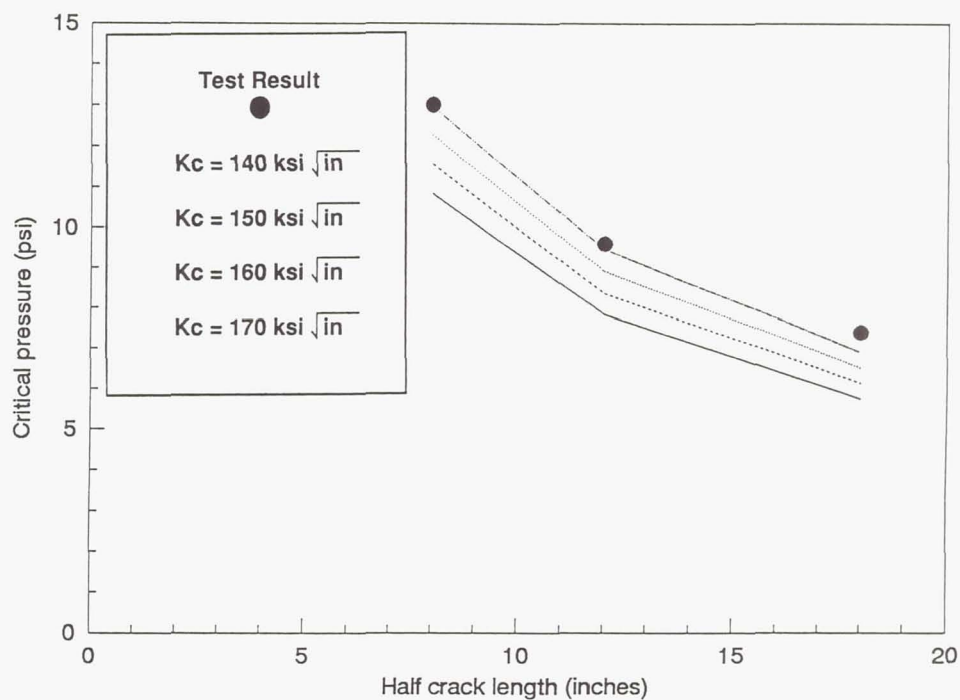


FIGURE 8. RESULTS OF ANALYSIS ASSUMING NON-LINEAR RIVET FLEXIBILITY



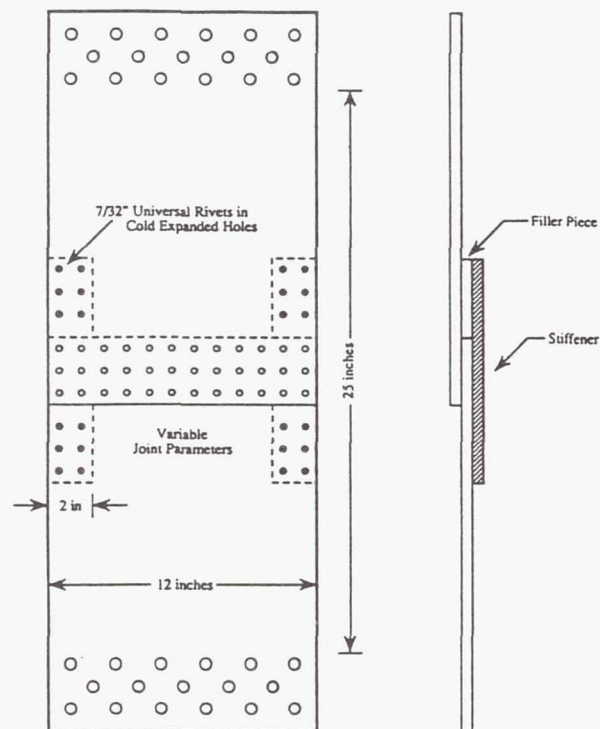


FIGURE 9. 12-INCH TEST PANEL

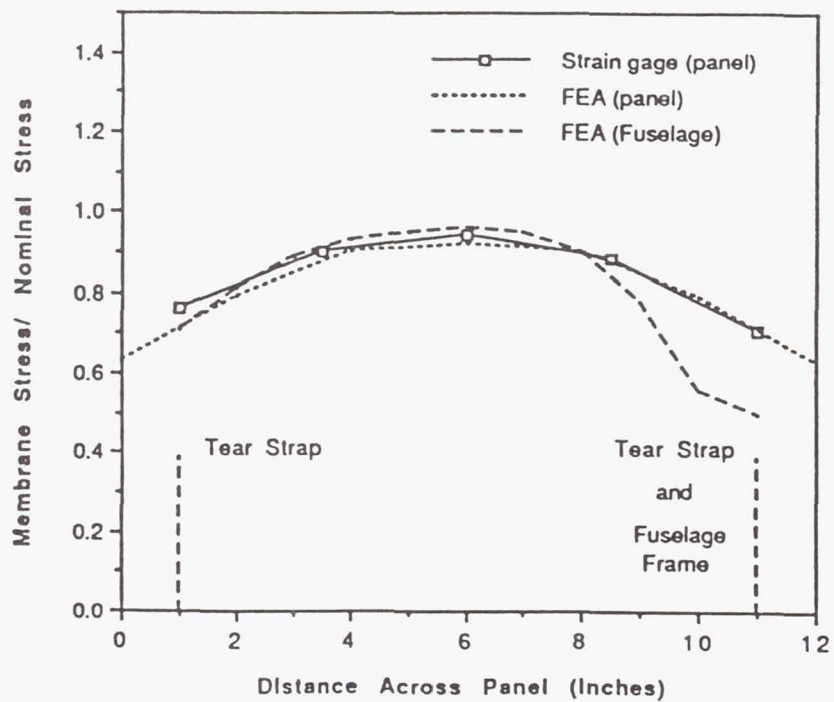


FIGURE 10. STRESS DISTRIBUTION IN 12-INCH PANEL

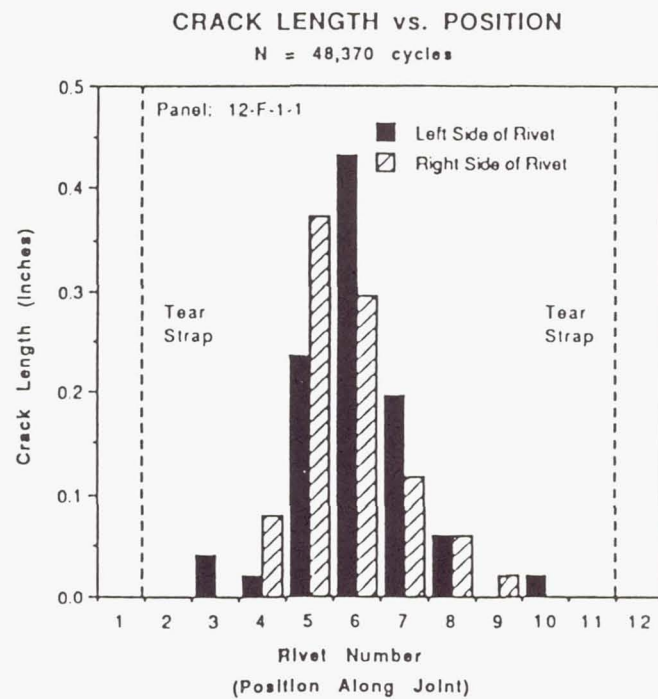


FIGURE 11A. CRACK PATTERN FROM 12-INCH PANEL TEST

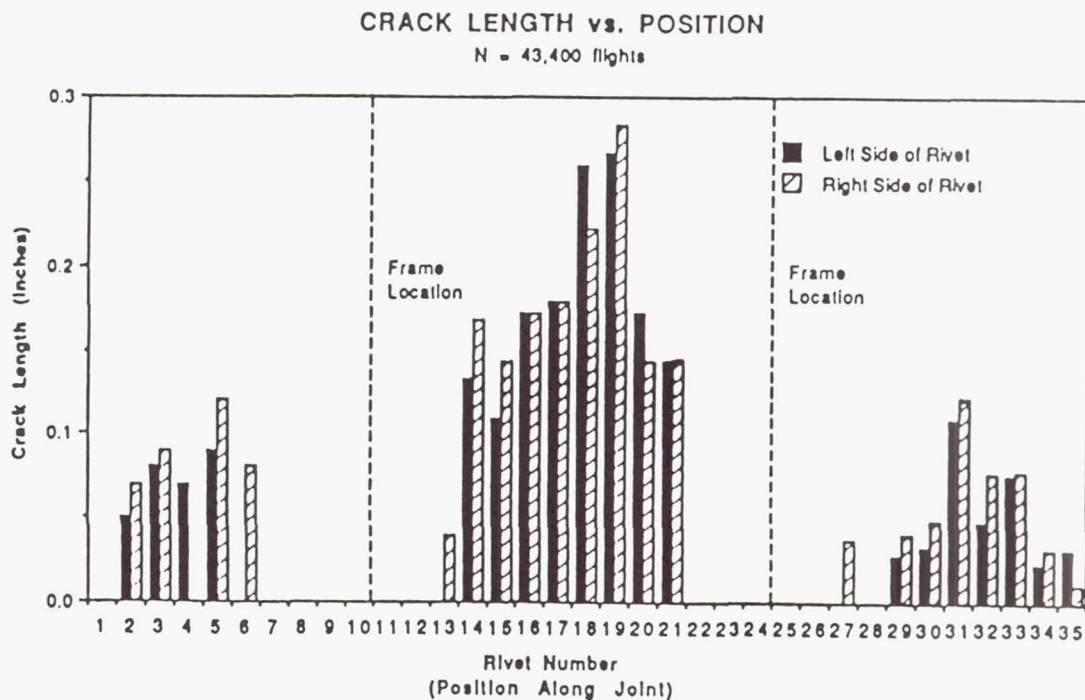


FIGURE 11B. CRACK PATTERN OBSERVED IN AN AGING AIRCRAFT

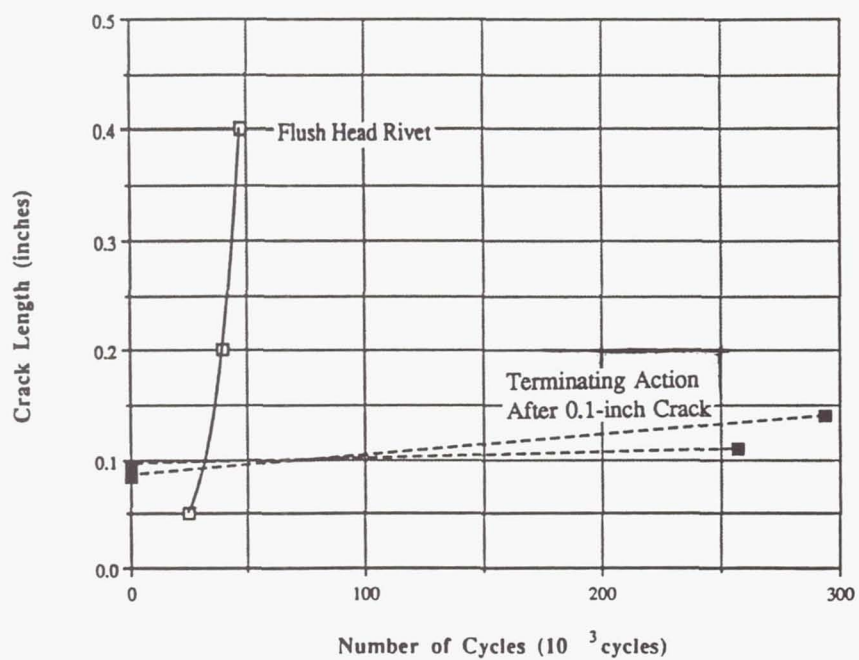


FIGURE 12. EFFECT OF TERMINATING ACTION ON CRACK GROWTH RATE

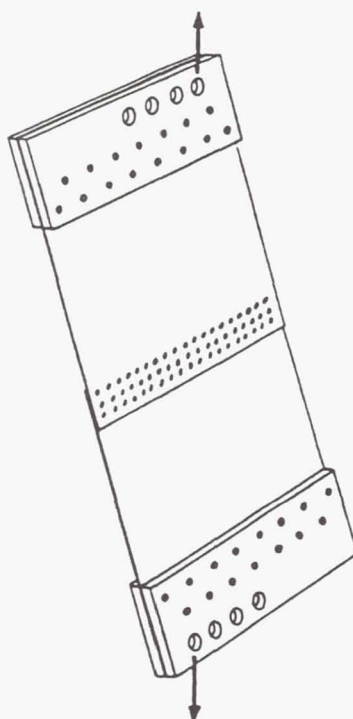


FIGURE 13. SCHEMATIC OF COMBINED SHEAR AND TENSION TEST PANEL



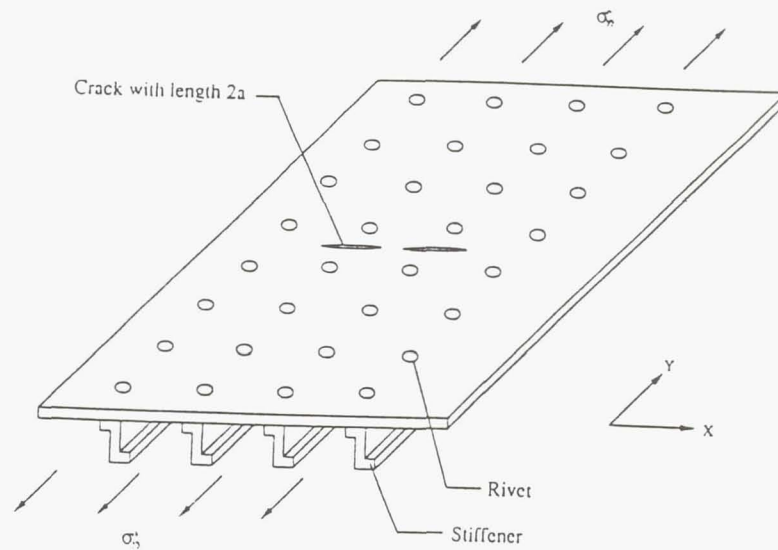


FIGURE 14. STIFFENED PANEL WITH TWO CRACKS

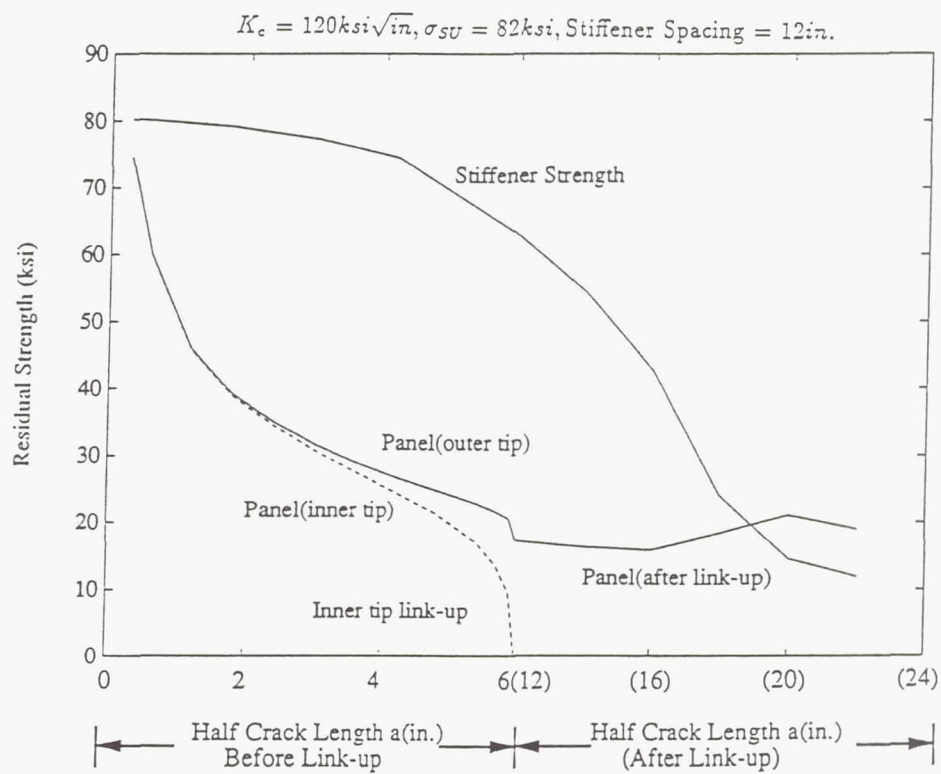


FIGURE 15. RESIDUAL STRENGTH OF THE STIFFENED PANEL

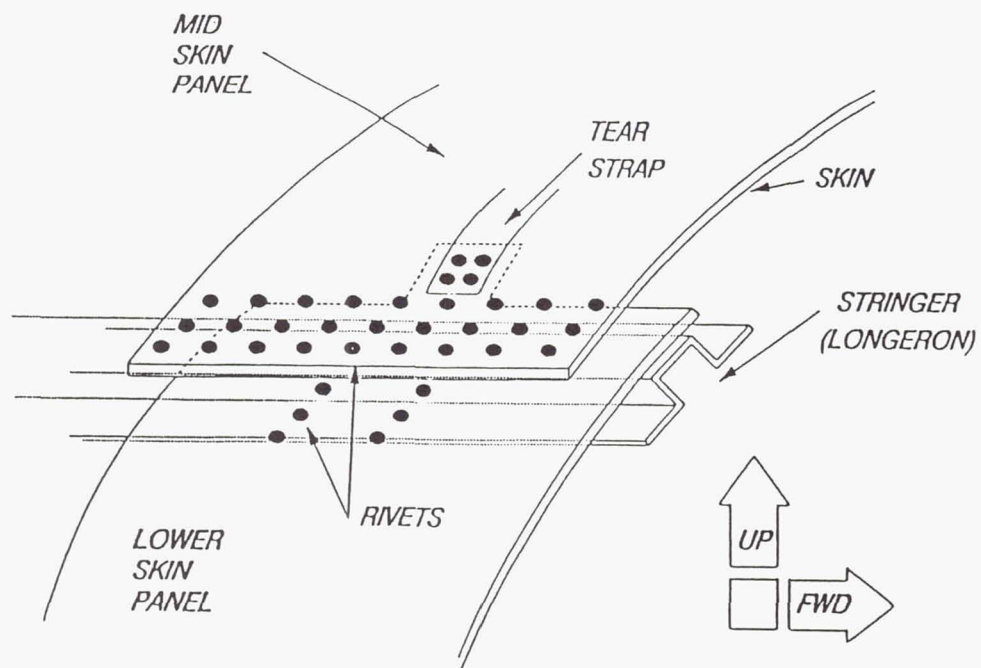


FIGURE 16. SKETCH OF TYPICAL RIVETED LAP-JOINTED FUSLEAGE PANELS  
(FROM REFERENCE [4])

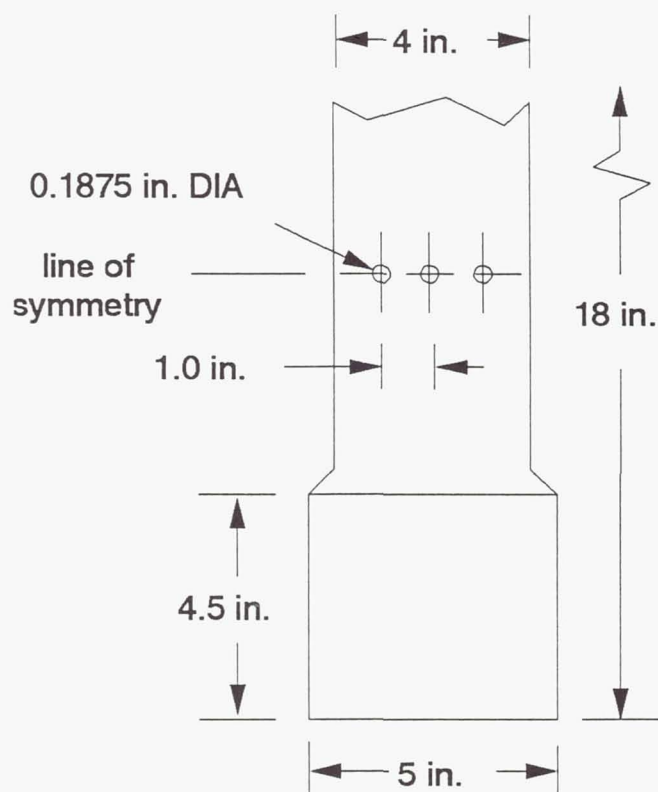


FIGURE 17. SCHEMATIC OF COUPON WITH SIMULATED MSD

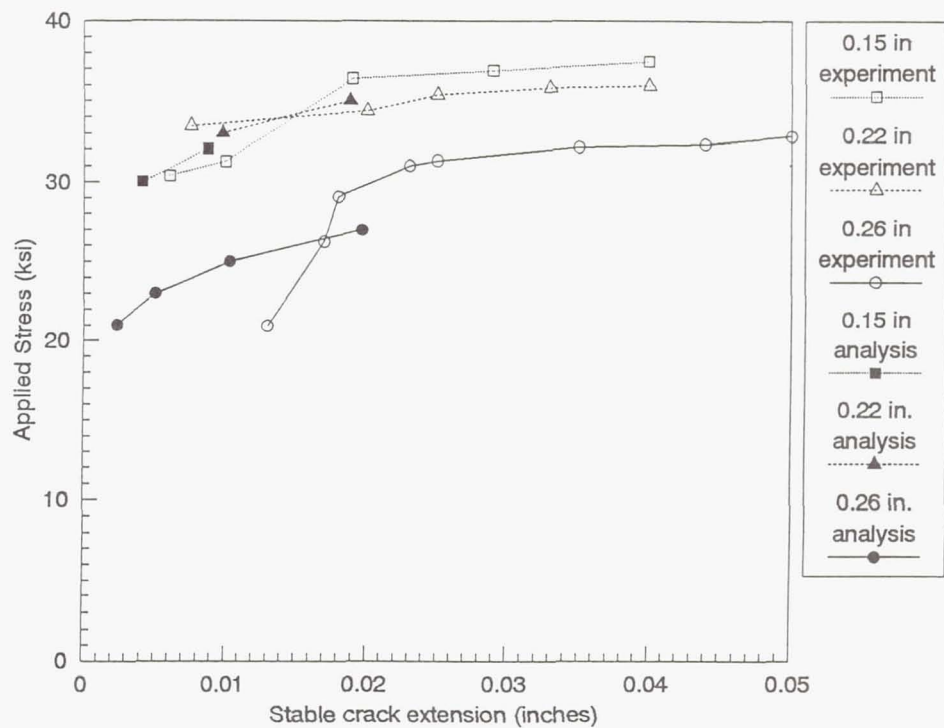


FIGURE 18. COMPARISON OF ANALYTICAL AND EXPERIMENTAL RESULTS FOR COUPONS.

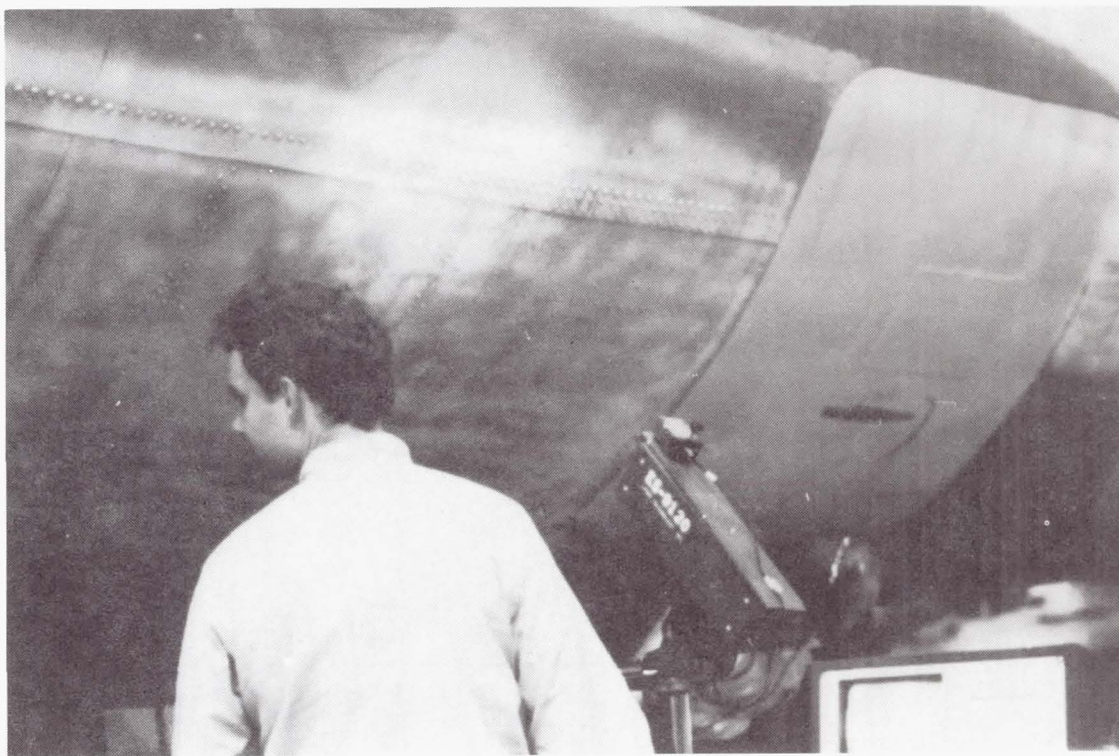


FIGURE 19. TRIPOD-MOUNTED SHEAROGRAPHIC INSPECTION SYSTEM



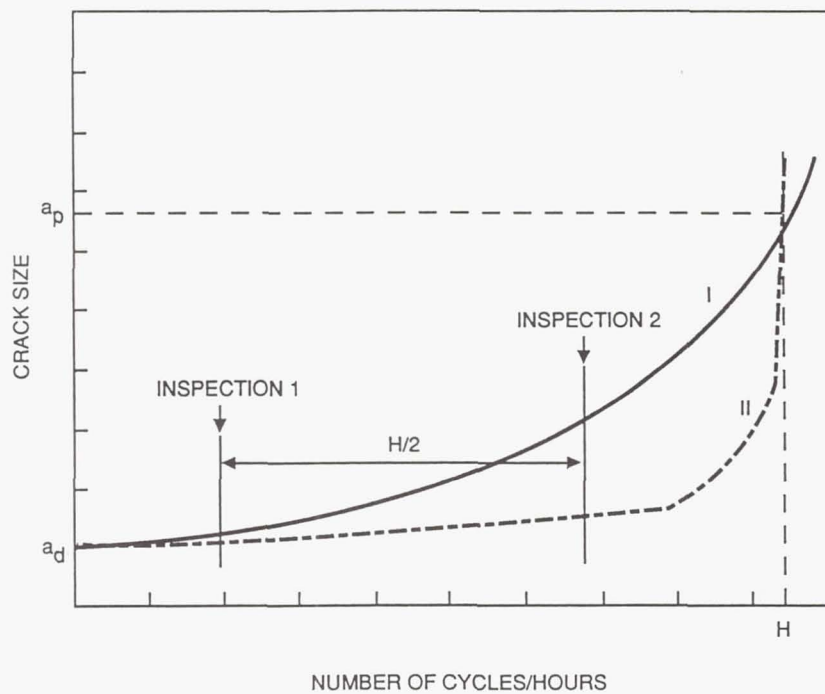


FIGURE 20. SIGNIFICANCE OF RELATION BETWEEN CRACK GROWTH AND INSPECTION INTERVAL (REFERENCE [7])

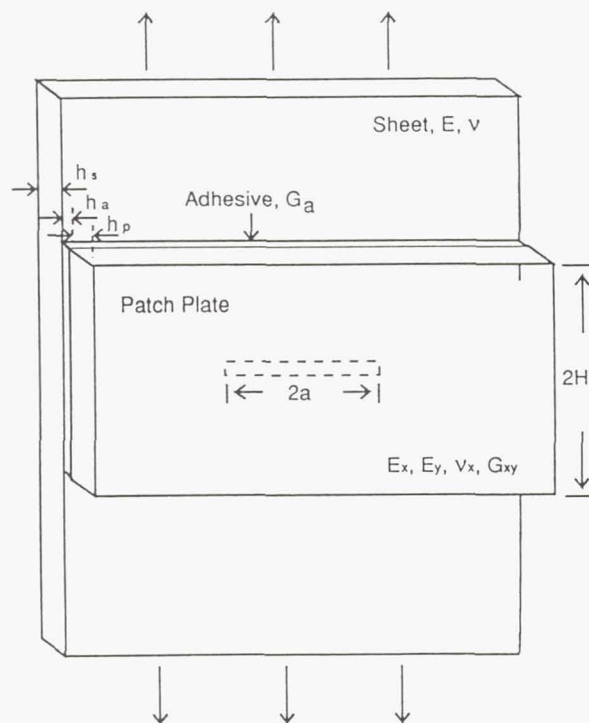


FIGURE 21. PATCH ON A CENTER-CRACKED PLATE (FROM REFERENCE [37])

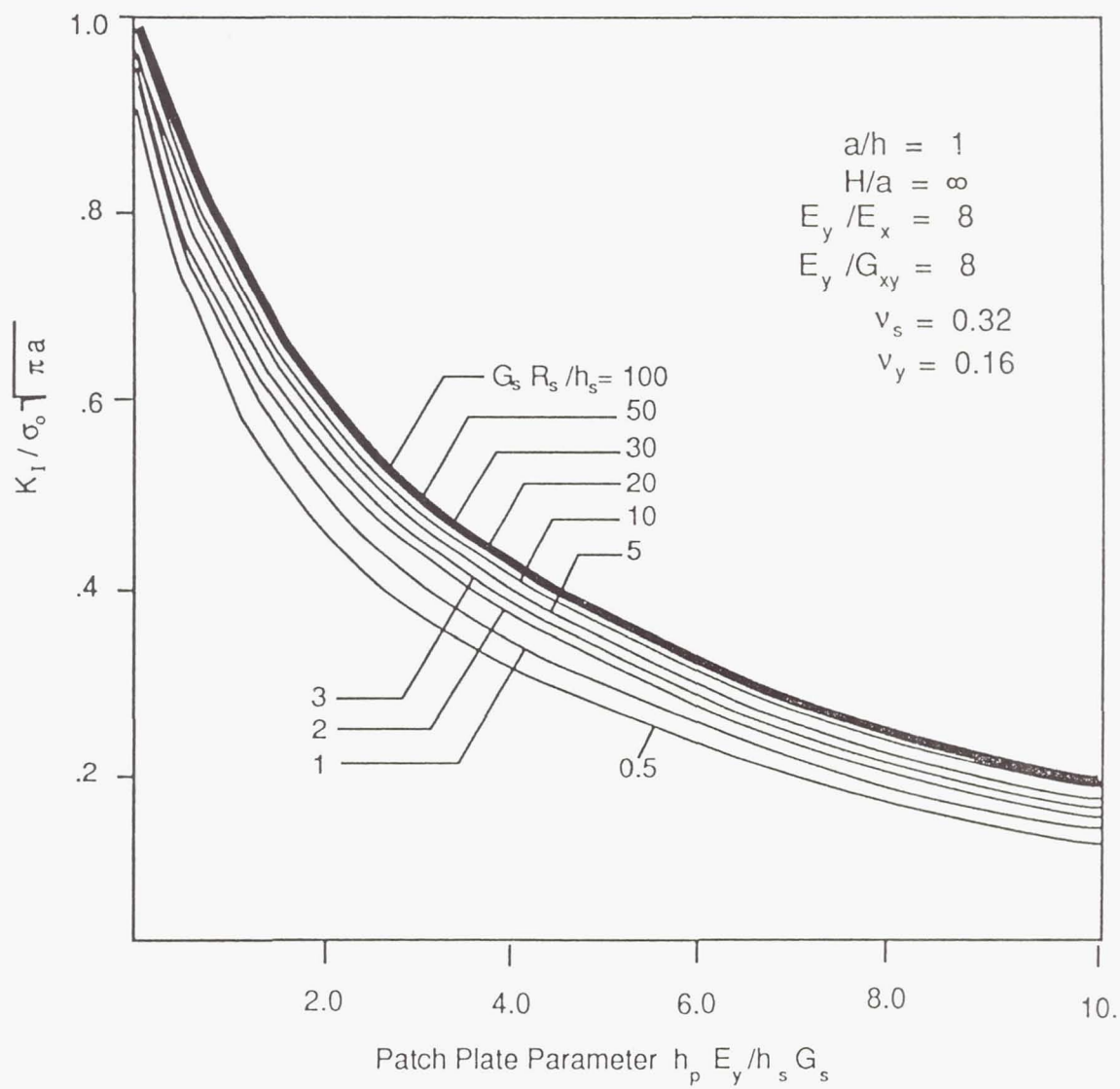


FIGURE 22 EFFECT OF PATCH PLATE ON THE STRESS INTENSITY FACTOR (FROM REFERENCE [37])

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## STATUS OF THE FAA FLIGHT LOADS MONITORING PROGRAM

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### ABSTRACT

In support of the Federal Aviation Administration Aging Aircraft Research Program, the Agency has established a Flight Loads Data Collection Program for Commercial Aircraft. The objectives of this Program are to:

- Review existing data collected by other sources including but not limited to U.S., Foreign, Military, etc.

- Collect current operational usage data from both large and small transport aircraft.

- Develop criteria for future generations of transports.

This paper presents the status of the various programs which are completed, underway or planned.

The FAA will be collecting, storing, and analyzing the data which characterize typical commercial transport operations. The airframe manufacturers will handle the task of calculating the loads and stresses.

### BACKGROUND

Many of today's large commercial transports and smaller regional/commuter carrier aircraft are being flown beyond their original intended service lifetimes, and the only feedback the FAA and the airframe manufacturers receive from current U.S. air carrier operations is the number of flight hours and landings which define the pressurization cycles and thus the major loads on the pressure hull. The stress and load history on the rest of the aircraft, i.e., wing, tail, flaps, controls, etc., which is dependent on flight operations, is assumed, but is largely unconfirmed.

Deregulation, advances in technology, and the increased demand for air travel have significantly modified how commercial aircraft are flown. Examples of these changes include:

- Hub and spoke scheduling concepts, replacing fixed routes.
- Noise restrictions affecting airport traffic patterns.
- Elimination of most circular holding patterns at major airports.
- Higher seat utilization (higher gross weight).
- Fly-by-wire/fly-by-light (FBW/FBL) control systems.
- Flight restrictions and airport curfews.
- Improved warning of turbulence.

Because the practice of flying both large and small commercial aircraft beyond their original intended service lifetimes is expected to continue into the foreseeable future, a need exists to acquire usage information to assure that the design criteria for future generations of aircraft are based on operationally relevant data.

The FAA Flight Loads Monitoring Program includes the following elements:

- The program in which researchers will collect, review and present summaries of significant data from earlier flight loads data collection and strain survey efforts. This will include available data from National, International and Military Transport programs.
- Installation and evaluation of an experimental optical disk recorder in the NASA owned Boeing 737.
- Development of data reduction routines and ground station operation.
- Installation and operational use of a prototype recording system on a Boeing 737 in U.S. domestic routine airline service.
- A pilot program for collecting loads data on 24 revenue transports of the major U.S. air carriers.
- A commuter category flight loads monitoring program.
- An in-flight strain survey which is being conducted on the FAA owned Beech King Air, in support of the commuter category flight loads monitoring program.

#### PURPOSE OF FLIGHT LOADS DATA COLLECTION PROGRAMS

The objectives of the flight loads research are to instrument regular in-service aircraft with state-of-the-art data acquisition equipment to collect new loads data: (a) to determine if the loading spectra being used or developed for design and test of both small and large aircraft are representative of actual operational usage, and (b) to develop structural design criteria for future generations of small and large aircraft.

Successful completion of this research will result in the existing loads data base being updated with increased accuracy from data collected using new state-of-the-art digital recorders. The original design and test spectra for the newer in-service aircraft will be validated. More severe routes will be identified affording airlines the opportunity of restricting the continuous use of a particular airplane in a severe environment.

#### FLIGHT LOADS MONITORING PROGRAM STATUS

##### Existing Data

Major organizations and government agencies have conducted data collection programs in the past. While many of these programs have objectives that are not directly related to aging aircraft, the recording technology and



applicable results will be identified and incorporated into this program. Flight Loads Data participants and research includes:

The Civil Aviation Authority Data Recording Program (CAADRP) which is primarily designed to collect data for pilot training. Data from British Airways aircraft are utilized for in-flight loads determination. Post flight analysis determines the occurrence of unusual events, some of which are used in a training program for pilot proficiency, and others are used to evaluate, for example, turbulence encounters.

The Netherlands National Research Laboratory (NRL) - Extensive studies on aircraft usage and load experience in relation to fatigue, and compilation of gust statistics were completed in 1989. Limitations in the program included the required use of the existing Aircraft Condition Monitoring System (ACMS) recorders, inability to change the aircraft operation, and cost constraints. In spite of this, extensive information was obtained for nearly 122,000 flight hours.

U.S. Navy Flight loads programs are used for the collection of maneuver statistics rather than the gust loads. Most military flight loads data collection programs are designed to determine inspection intervals in fleet aircraft. Their procedures do not support civil programs because of the different intended results and operating environment, however the equipment used to acquire the necessary information may have application in civil use.

The current FAA flight loads monitoring program includes a separate task to produce an International Flight Loads Data Bank. This task will:

- Work with the commercial airframe industry to determine the priority, needs and organization of the material.
- Identify overseas sources of flight loads data for both large transport and regional/commuter aircraft determined to be of significant value in furthering the aims of the FAA's developing program.
- Acquire and publish loads data identified above.
- Document fully the means used to acquire the data, giving the rationale for the selection of the variables recorded, sampling rates, and a description of the sensors, transducers, gauges, etc., used to develop the raw signals and their characteristics.
- Document the methods used to analyze and organize the data, filters, algorithms, statistical methods.

This task is being handled under an agreement on cooperation between the FAA and the Netherlands Civil Airworthiness Authority. Most of the work will be done in Europe.

### Large Transports

Many of the parameters currently recorded as time histories on the existing large transport Digital Flight Data Recorder (DFDR) 25 hour loop tape will be routed to a special high density optical disk recorder which is

capable of collecting and storing many months of flight usage information. Data from these recorders will be analyzed on the ground and periodically presented as statistical summaries in FAA Technical Reports. The complete system of onboard recorder data transfer and ground station operation is being thoroughly checked out prior to installation of the first recorder to be used during revenue operations.

An experimental optical disk recorder has been installed in the NASA owned B-737 aircraft. This system is currently collecting complete time histories of selected flights of the same data being stored in the DFDR (Crash Recorder). Analysis of these data is expected to provide the following: (a) Reliability of the optical disk recorder versus solid state recording instrumentation, (b) Comparison of data transfer rates between the optical system and a solid state system, (c) A measure of the data quality of today's advanced digital data recording systems, (d) Flight data from which data health monitoring algorithms can be derived, coded and tested, (e) Flight usage data from which flight characterization data reduction algorithms can be coded and verified, i.e., Nz exceedances, altitude, airspeed bands, etc.

The FAA is working closely with the airlines to obtain volunteers to host the recorders and provide other data as required. The anticipated onboard data acquisition configuration is described on Figure 1. The notations 429, 573, 591 and 717 refer to on-board Aeronautical Radio, Inc., (ARINC) data bus identification numbers.

Two certified Sundstrand optical disk recorders have been purchased. One will be installed on a U.S. air carrier Boeing 737 and the second used as part of the ground station. Data will be collected, transferred and analyzed by the ground station.

Twenty-four recorders will be installed in various airplane models over four years at the rate of six per year. Data will be collected while operating under the U.S. air carrier rules of Federal Aviation Regulation (FAR) Part 121 in normal passenger operations. The data flow will be as indicated on Figure 2.

Data will be off-loaded and reviewed periodically at approximately "B" level maintenance intervals. Measured normal and lateral acceleration data will be filtered to separate gust and maneuver events per Figure 3. The maneuver event summaries will be used directly by the manufacturers to develop component loads using the validated loads programs. Filtered gust data will initially be defined as high frequency accelerations measured in service. Through the use of airplane aerodynamic parameters, derived gust velocity spectra will be developed and provided in spectrum format to the manufacturer.

The manufacturer will then use both the maneuver and gust information to calculate component loads, and with the validated stress analysis program and detailed design data, develop the individual component stresses. See Figure 4. The procedure which the manufacturers use to develop reliable airplane and component loads from the relatively simple measured parameters, using their own validated loads and stress analysis programs, is described in detail in Reference 1.



The ultra high density optical disk recorder is capable of storing up to 300 megabytes (600 using both sides) of complete time history flight data representing 600-800 flight hours of normal operation transport service.

The ground station data reduction software will have the ability to recognize the aircraft situation (take-off, climb, cruise, etc.,) and based on this situation select the appropriate data reduction and storage algorithms. The recorded data will be processed, separated, and stored on two permanent storage records: (a) typical aircraft usage characteristics, and (b) individual flight summaries.

As part of the structural life assessment process, the large transport airframe manufacturers conduct a highly detailed component by component damage tolerance analysis.

For a major structural member such as a wing, fuselage or tail, validated equations exist for computing the Bending Moment (M), Torsion (T), and Shear (V) for the wide variety of anticipated structural load conditions.

The stress at any particular location can therefore be estimated by yet another validated relationship:  $\text{Stress} = f(M, T, V)$ , Figure 5, and the stress history (or anticipated stress history) for the component can be defined.

The problem in large transport commercial aviation is that the design parameters used for the fatigue and damage tolerance analyses were derived from data collected from the early 1950's and 1960's, which were obtained with sometimes questionable data acquisition and reduction processes.

#### Small Transports

Manufacturers and operators of small transports have indicated a strong interest in monitoring loads parameters. This has resulted in the need to conduct research in the following specific areas:

- Development of a fleet tracking system to provide fleet usage information and develop trend data.
- Development of reliable lateral load spectra for empennage service life determination.
- Possible development of a certified lightweight, low cost airframe strain or fatigue recorder.
- Development of inspection schedules which realistically reflect typical usage.

The regional/commuter effort will involve the instrumentation of aircraft with gross weights below 50,000 pounds, and will be propeller driven. Since both small and large transports will be required to be equipped with Digital Flight Data Acquisition Units (DFDAU), the flight usage data for the aircraft of 10 seats or more can be acquired in a means similar to the larger transports. Acquiring the data from the smaller airplanes having less than 10 seats operating under FAR Part 135 will involve the installation of special



recorders, instruments, and sensors, and will limit the ability to transfer the equipment from one FAR Part 135 airplane to another.

The analysis and interpretation of the commuter data will include consideration of both geographic area effects and seasonal fluctuations. The latter will necessitate acquiring the aircraft usage information in yearly cycles in order to properly assess the seasonal variations. In addition, the effects of yaw damper and autopilot will be studied. A detailed literature search of both domestic and international service usage will also be conducted.

The development of a flight loads parameter recording program for regional/commuter aircraft will in many cases require additional wiring to acquire all the necessary air data signals and controls information for post flight data analysis. While the atmospheric gust environment for commuters is the same as for large transports, the small aircraft usage (i.e., length of flight, cruise altitude, etc.,) and hence the exposure is quite different. In addition, it is hypothesized that the maneuver loads environment may also be different since commuters routinely use alternate runways, and fly on different flight paths. The commuter flight loads monitoring program will be developed based on knowledge gained during the initial planning and implementation phases of the large transport flight loads monitoring program. Additional steps are considered necessary in the implementation of the small transport flight loads monitoring program.

The development of empennage lateral loading criteria, and the relationship of exposure to lateral and vertical gust velocity will form part of the research activity. Using relatively simple parameters such as lateral acceleration at the airplane c.g. and airplane weight and speed may not be sufficient for the definition of a fin and aft body lateral loading spectrum. Furthermore, some c.g. vertical acceleration and speed data have been collected on small airplanes, and it is not known if this can be used to predict a lateral loading spectrum. In order to understand these relationships, a limited flight strain survey program will be conducted on the FAA owned Beech King Air. Strain gauges, surface position transducers and accelerometers will be installed on the airplane and the relationships determined. This limited flight test program will also be used to determine the minimum number of additional sensors and their location required on the operational flight loads monitoring program.

## TECHNICAL DETAILS

### Airborne Flight Loads Data Recorder Hardware

The new technology Sundstrand optical disk recorders can be installed in all large transports. This recorder consists of a high performance ultra high density erasable optical disk recording system with the capability to: (a) interface with the large transport aircraft ARINC 429 data bus, flight data entry panel, CG accelerometer, and other discrete and analog inputs, (b) provide a significant amount of embedded computational capacity for data acquisition, editing, and parameter selection, (c) be reprogrammable with the capability to accommodate future DFDAU enhancements, and (d) provide sufficient non-volatile memory (minimum of 300 megabytes) for extended aircraft operation without down loading more frequently than the standard "B"

Level Maintenance check of eight to ten weeks. See Figure 6, Sundstrand Recorder.

The recorder will utilize the auxiliary output of the newer dual microprocessor DFDAUs. Microprocessor #1 is dedicated to providing the required Digital Flight Data Recorder (crash recorder) input. Microprocessor #2 is optional equipment which is fault isolated from microprocessor #1, and is beginning to attain wider use as an operator maintenance tool. See Figure 7.

This new recorder will have the ability to record complete time histories of over 40 flight, control, and configuration parameters, and record and store between 500 and 800 flight hours of the data in 12 bit words. The new system will allow for extensive data quality editing, thus assuring that the data used in the subsequent analyses will be of the highest quality.

State-of-the-art digital onboard microprocessor based recorders ("Smart Recorders") are being used extensively by the military and have eliminated nearly all the tedious ground processing required by most previous methods. These recorders can use either bubble memory or be installed with a hard drive memory for post flight data interrogation and analysis and will likely be used in the regional/commuter program. Three types of data parameters are usually obtained with this system.

- Peaking parameters. The accelerations and rates (pitch, roll, etc.), are in this category. The criteria defining these need to be specific and coded before they can be identified. Peak and valley determination is a form of data compression, such that each time a peak or valley determination of a parameter is made, the instantaneous value of numerous other parameters is also recorded. The separation of gust and maneuver normal and lateral accelerations can also be accomplished onboard.

- Time parameters such as airspeed and altitude which can be recorded whenever these parameters cross predetermined range boundaries.

- Discrete channels such as flap position, take-off and landing switches, squat switch, yaw damper, autopilot and others which trigger the recording of data whenever they change state.

These recorders can typically store 2 megabytes of compressed data, which is sufficient to store data between "B" Level Maintenance checks.

#### Ground Station

The new flight loads data recording ground station will be a state-of-the-art microcomputer with the capability of analyzing and making decisions on data selection and permanent storage for future flight data evaluations. In order to expedite the data transcription process (i.e., optical disk to ground based computer), the ground station will input the airborne optical disk data using an off-the-shelf hardware Standard Computer System Interface (SCSI). Data will be transcribed and stored in the ground station central memory, and software will be written to recognize the aircraft situation (climb, cruise, etc.), and based on this situation, select the appropriate data reduction and storage algorithms. When the data analysis is complete, databases consisting of individual flight summaries, and complete time histories will be compressed



and stored on two sided 600 megabyte optical disks. It is estimated that downloading the optical disk to the ground station central memory will require no more than 30 minutes. The initial design of the aircraft and ground station interface is presented on Figure 8. The commuter ground station will have a similar design but will not involve the optical disk.

### Data Health Monitoring

This is the FAA's initial digital flight loads data collection program, and is expected to be in existence for many years. The hardware will be transferable among most large domestic transports and operators, and a number of automated digital data processing procedures will need to be developed and incorporated into the total data reduction and analysis process. Since, at this time, no data are available from the new dual microprocessor DFDAUs, it is not possible to assess the data condition and develop and incorporate detailed data editing algorithms, however the concepts which will be used are presented and discussed. The two principal sources of data editing procedures are presented in references 2 and 3. In addition to the built-in bit tests within the optical disk recorder, the data health monitoring will consider several issues including (a) activity testing to verify that the data received from the various data channels are exhibiting at least a minimum of reasonable activity, (b) wild point editing to remove and replace data spikes and, (c) range tests to ensure that the data processed are within reasonable upper and lower limits, and represent information from a realistic physical environment.

The health monitoring represents a major endeavor which, in itself, does not produce new design criteria or verify existing criteria, however, the lack of a reliable and consistent data health monitoring system could compromise the results of this entire program both now and in future years.

### Gust and Maneuver Acceleration Separation

Inspection of many power spectra of the center of gravity normal acceleration data indicates that the lower frequency maneuver accelerations are sufficiently far removed from the gust responses that suitable digital filters can be used to reliably separate gust events from maneuver events. Examples of such power spectra appear on Figure 9, where the peaks occurring below 0.1 Hz are due to pilot induced maneuvers, while those between 0.1 and 1.0 Hz were identified as being due to turbulence. Peaks above 1.0 Hz are thought to be due to wing first bending mode.

Accordingly, the filters used in this program, illustrated in Figure 10 and described in reference 3, were developed based on the methods of reference 4, and utilized herein to separate pilot induced accelerations from aircraft gust response.

Results of the application of these filters to a typical time history are presented in Figure 11.

### Derived Gust Velocities

Derived gust velocities ( $U_{de}$ ) are computed using the method described in reference 5, and use the resultant gust normal accelerations from the gust and maneuver separation process.



where

$$U_{de} = 2Wa_n / m\rho_o S V_e K$$

$U_{de}$  = derived (effective) gust velocity (fps)

$a_n$  = airplane non-dimensional normal acceleration in g units

$m$  = lift-curve slope, per radian

$\rho_o$  = air density sea level, slugs/cu ft

$S$  = wing area, sq ft

$V_e$  = equivalent airspeed, fps

$W$  = airplane gross weight (lbs)

$K$  = dimensionless "alleviation factor"

In this program, the lift-curve slope is the untrimmed flexible lift-curve slope for the entire airplane, and is a function of Mach number, altitude, and flap deflection. Since the new data recorders will be recording complete time histories of all measured parameters, the corresponding time histories of derived gust velocities will be available. These time histories will then be compacted into peak/valley exceedances using level crossing techniques, and frequency histograms of  $U_{de}$  will be plotted in pressure altitude bands (e.g., 5,000 ft, 10,000 ft, etc.)

#### Collection and Analysis of Data

All of the flight data from these new and improved recorders will be analyzed and then permanently stored on the double sided high density optical disks with maximum storage capacity of 600 megabytes. A flight usage database will be established and updated with the continuous supply of additional commercial aircraft recorder data. Analysis of subject data is expected to provide airframe manufacturers with information with which to assess the structural (fleet-wide) usage severity of aircraft, structural components, and control surfaces loading spectra, etc. In addition, the individual flight usage summaries will provide both operators and manufacturers with (a) trend data such as operator usage differences attributable to route structure, (b) data to assess how well the original design criteria are reflected in the current typical fleet service usage and (c) data to establish structural design criteria for future generations of aircraft.

The new flight loads data analysis will be more comprehensive than any prior commercial flight data collection endeavor, and will provide user friendly instructions which automatically separate the output into individual flight summaries and typical aircraft cumulative usage characteristics.

#### Flight Phase Descriptions

The analysis of the cumulative usage data will involve separating the data into the following phases of flight: taxi, take-off, departure, climb, cruise, descent, approach, and landing. (Figures 12 and 13.)

## Individual Flight Summaries

Algorithms will be developed to query the time history from each flight to identify and store the salient facts about each flight, to be later printed and used in the development of statistical summaries of extreme values or typical individual flight usage characteristics. The October 1991 FAA requirements for additional flight parameters to be recorded on the crash recorder will provide additional data from which automated statistical summaries, not available from prior flight load data collection efforts, can be developed. A sample of the typical individual flight data which will be collected and stored will include take-off and landing gross and fuel weight, maximum and minimum normal and lateral acceleration by flight phase, maximum control surface deflections, and airspeed and altitude of each flap extension and retraction. Periodic reports summarizing these results will be provided which can be typified by the following:

### Flight Summaries

<u>Flight Segment</u>	<u>Flap Position</u>	<u>Altitude Range</u>	<u>Airspeed Range</u>	<u>Nz Range</u>	<u>Ny Range</u>
Departure	#1				
Departure	#2				
Departure	#3				
Departure	::				
Departure	::				
Departure	#N				
Climb					
Cruise					
Descent					
Approach	#1				
Approach	#2				
Approach	#3				
Approach	#::				
Approach	#::				
Approach	#N				

### Takeoff/Landing Data Summary

The new FAR Part 121 crash recorder requirements include ground velocity as a required parameter, and this could be used to calculate and present a number of new takeoff and landing parameters. A data file of these calculated takeoff and landing data records could be established.

### Model/Survey-wide Statistical Flight Usage Summaries

The complete time history data collected in these surveys will be summarized by aircraft model, analyzed, and presented in periodic FAA reports. A generic summary and typical statistical summaries are presented on Figure 14.

### Flight Profile Statistics

Typical flight profile statistics will include: Distributions of flight length, airspeeds, and altitudes, time on auto pilot status, altitude,

airspeed, gross weight, and fuel weight when changing flight phases (cruise, climb, etc.), distributions of take-off and landing gross weight and fuel weight. See Figure 15.

In addition to Figures 14 and 15, the following are typical examples of other data which will be made available from the individual flight summaries data bank;

- Level flight airspeed and altitude distribution.
- Gross weight distribution.
- Distribution of flight duration.

#### Acceleration Derived Statistics

For each of the flight segments, distributions of normal and lateral acceleration will be presented. See Figure 16 for a sample for vertical acceleration from reference 6. For each of the flight phases and for each altitude band, normal (vertical) acceleration will be separated and reported as gusts, maneuvers and total normal accelerations. From the gust portion of the normal accelerations, peak and valley derived gust velocities will be calculated and sorted by altitude and flap position. See sample normal acceleration on Figure 17 and derived gust velocity on Figure 18. Lateral acceleration will be reported as totals only.

#### Control Surface and Configuration Data

The control surface and configuration data summaries in this new program will be the most comprehensive of any survey conducted on the civil fleet thus far. These summaries will include control position profiles for Rudder, Aileron, Elevator, Stabilizer, and Spoiler for each of the flight phases. A typical profile is presented on Figure 19.

The flaps and high lift device deflection analysis will be handled in a somewhat similar manner, as shown in the example on Figure 19. For the take-off, departure, approach, and landing phases, the altitude and airspeed for each flap extension and retraction will be stored. From these, probability plots of dynamic pressure will be created for each flap setting during both extension and retraction. The distribution of airspeed during both extension and retraction will then be plotted with airspeed limit for that model and flap position as appears on Figure 20.

#### SUMMARY

The various elements of the FAA Flight Loads Monitoring Program for both large and small transports are in place.

An agreement to assemble existing international transport flight loads data is in work between the FAA and NLR. Tasks are being written for the Wichita State University National Institute for Aviation Safety to support the commuter flight loads program.

A new onboard technology flight loads data recorder has been selected and delivered for evaluation on the NASA B-737, and a prototype flight loads recording system is scheduled to be installed in a B-737 aircraft during 1991.



Six additional recording systems are planned to be installed on other transport aircraft in each of the next four years. When this new data collection system for large transports is operational with 24 recorders, 75,000 flight hours of data are expected to be collected and processed yearly. The new state-of-the-art recorders, automated data reduction routines, and comprehensive data analyses will result in a highly efficient, durable, accurate, and flexible system, capable of monitoring large quantities of data for typical usage of both current in-service aircraft and new aircraft models as they enter routine fleet service.

A limited flight test strain program is being conducted on the FAA Beech King Air to determine what parameters need to be measured for the commuter flight loads program. Depending on the results of the initial phase of the flight test program, it may be necessary to record stresses directly to obtain the desired data on small transports in commuter operations.

Further expansion of the program, into monitoring foreign aircraft operating under U.S. Registry, or U.S. manufactured aircraft operating abroad, will be considered following an initial evaluation of data collected during the pilot transport and commuter monitoring programs.

## REFERENCES

1. AIAA 91-0258, The New FAA Flight Loads Monitoring Program, Presented at the 29th Aerospace Sciences Meeting, January 7-10, 1991, Reno, Nevada, Barnes, Terence and DeFiore, Thomas, FAA.
2. DOT/FAA-CT-89/36-1, NASA Contractor Report 181909, Volume 1, The NASA Digital VGH Program-Exploration of Methods and Final Results, December 1989.
3. Systems Development Corporation Integrated Services, Inc., Company Document PDD-79-01, Program Description Document for Automated Edit Program for Digital VGH Data Analysis, August 1977.
4. NASA TRR 179, Determination and Analysis of Numerical Smoothing Weights, Graham, R. J., 1963.
5. NACA Report 1206, A Revised Gust-Load Formula and Re-Evaluation of V-G Data Taken on Civil Transport Airplanes From 1933 to 1950, Pratt, Kermit, and Walter Walker, 1954.
6. FAA-RD-71-69, Airline Operational Data from Unusual Events Recording Systems in 707, 727, and 737 Aircraft, August 1971.

Figure 1  
U.S. AIR CARRIER ON-BOARD LOADS DATA ACQUISITION

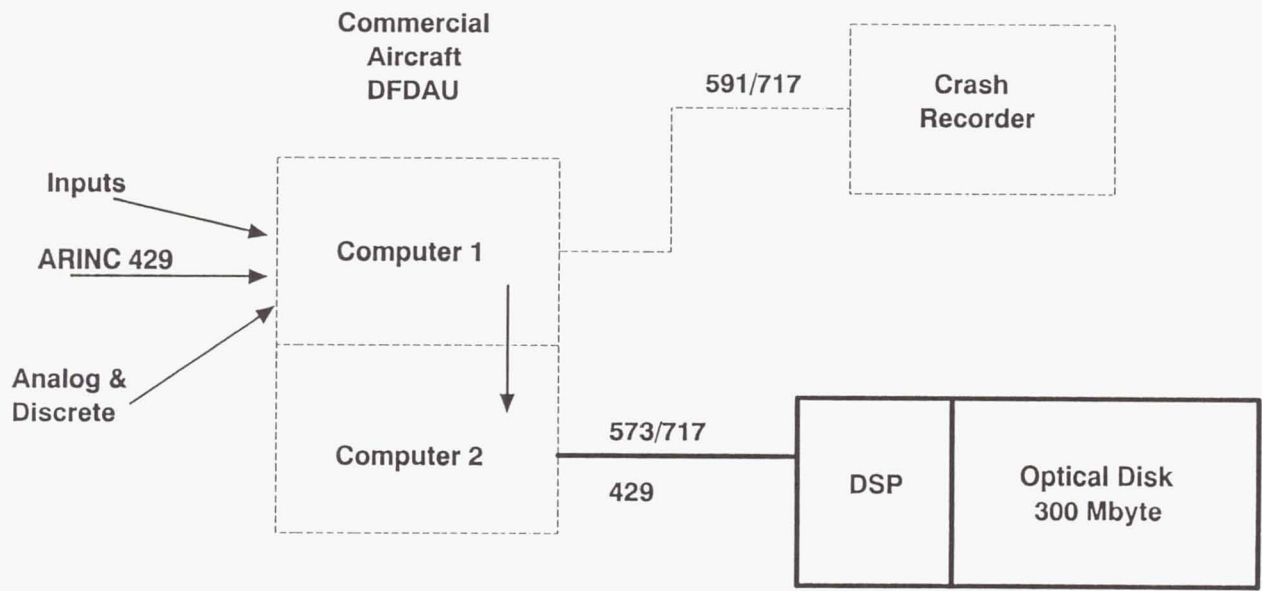


Figure 2  
COLLECTION DATA FLOW - STEP 1

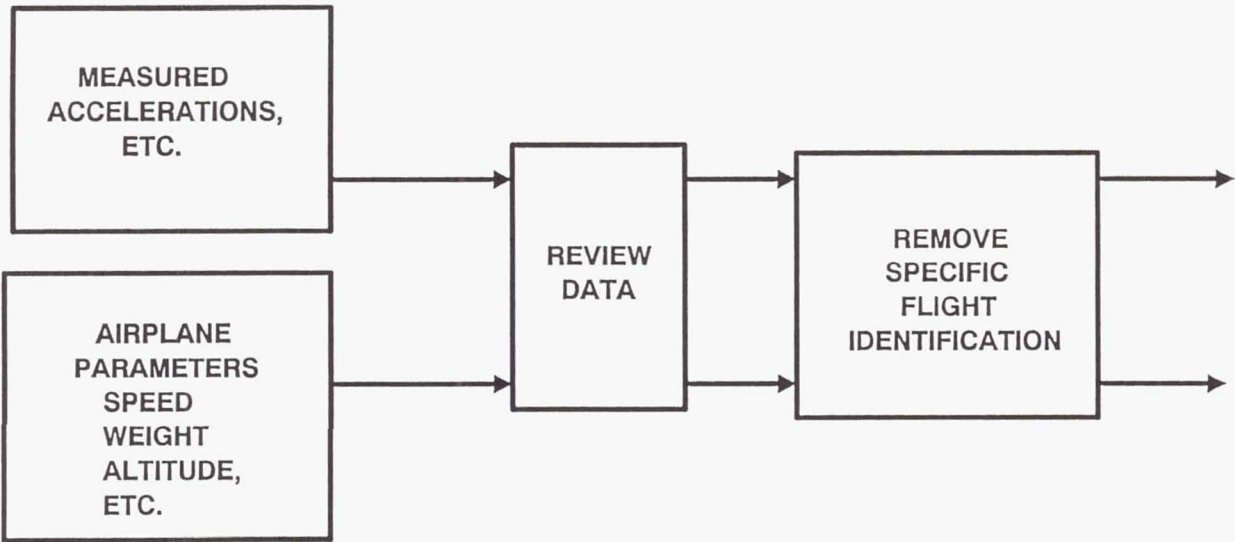




Figure 3  
COLLECTION DATA FLOW - STEP 2

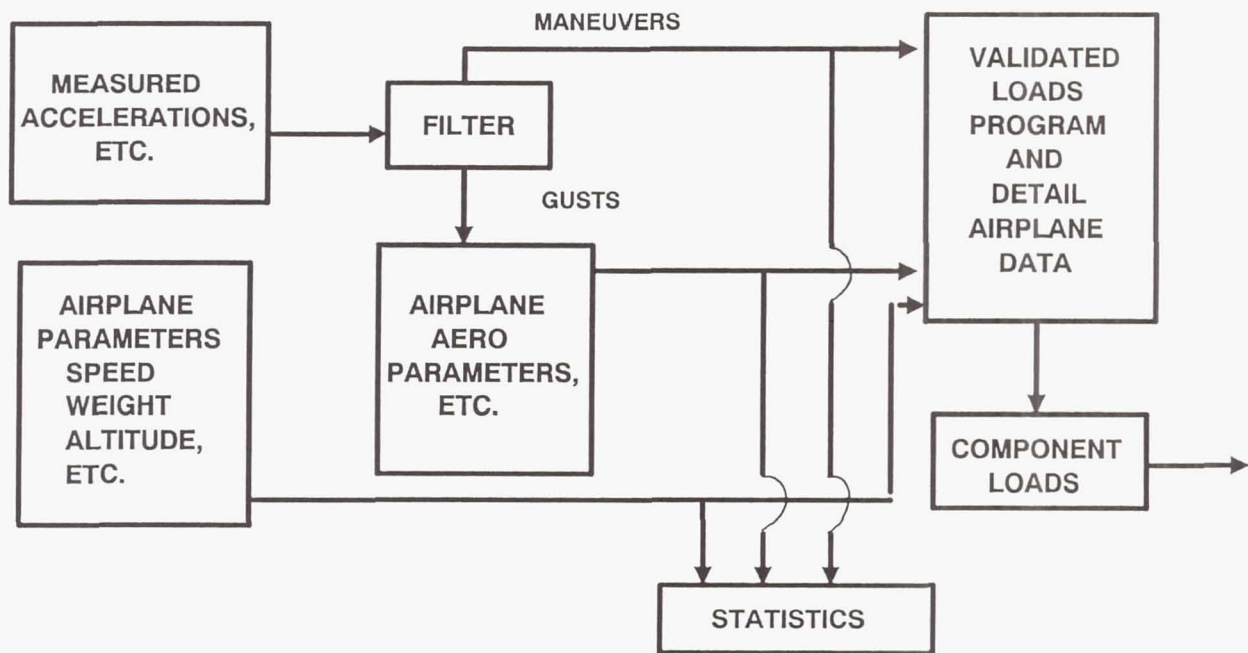


Figure 4  
COLLECTION DATA FLOW - STEP 3

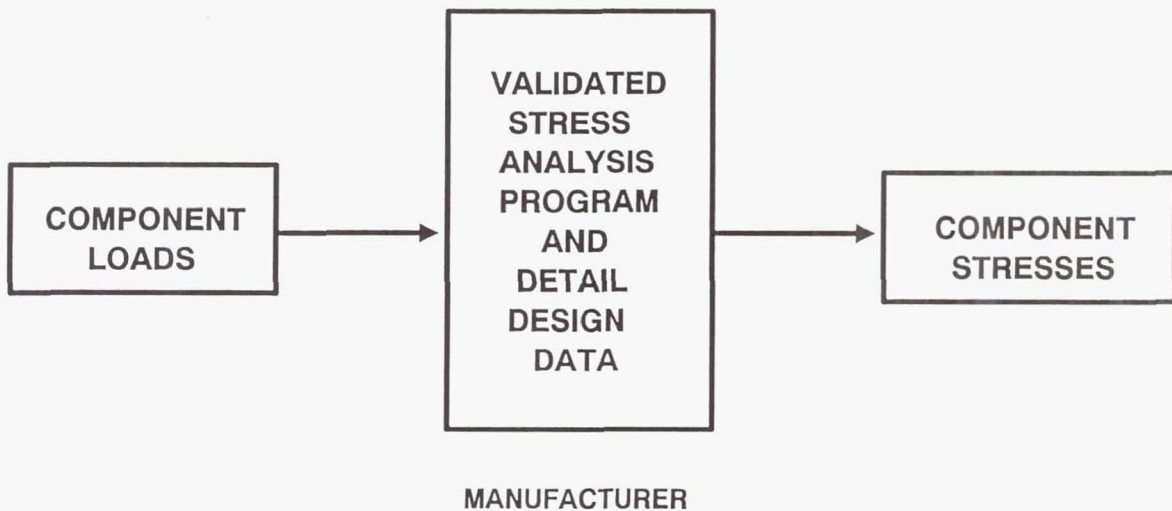


Figure 5

## FLIGHT LOADS - FATIGUE RELATIONSHIPS

FOR EACH AIRFRAME STRUCTURAL COMPONENT, THE FOLLOWING EXISTS:

BENDING MOMENT (M)	= F (DESIGN PARAMETERS)	VALIDATED
TORSION (T)	= F (DESIGN PARAMETERS)	VALIDATED
SHEAR (V)	= F (DESIGN PARAMETERS)	VALIDATED
STRESS	= F (M, T, V)	VALIDATED
FATIGUE	= F (STRESS)	

PROBLEM:

DESIGN PARAMETERS = F (1950's, 1960's USAGE)

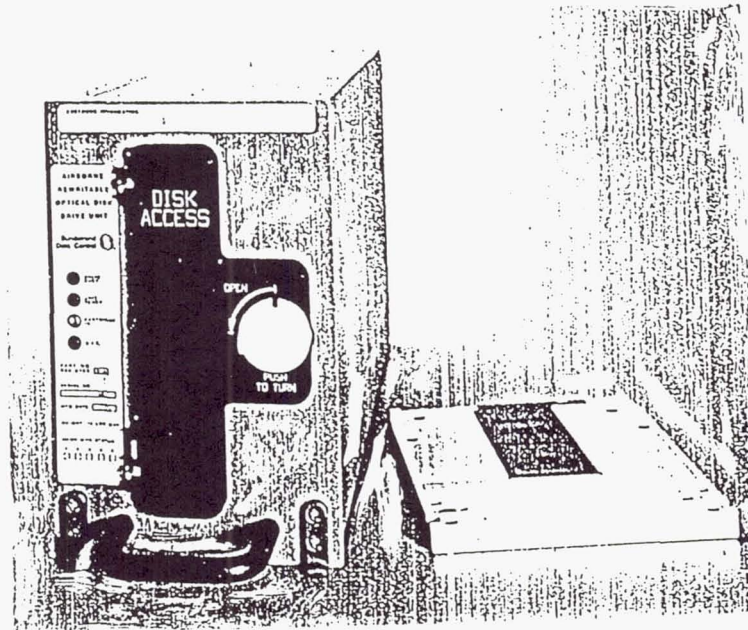


Figure 6

## SUNDSTRAND RECORDER

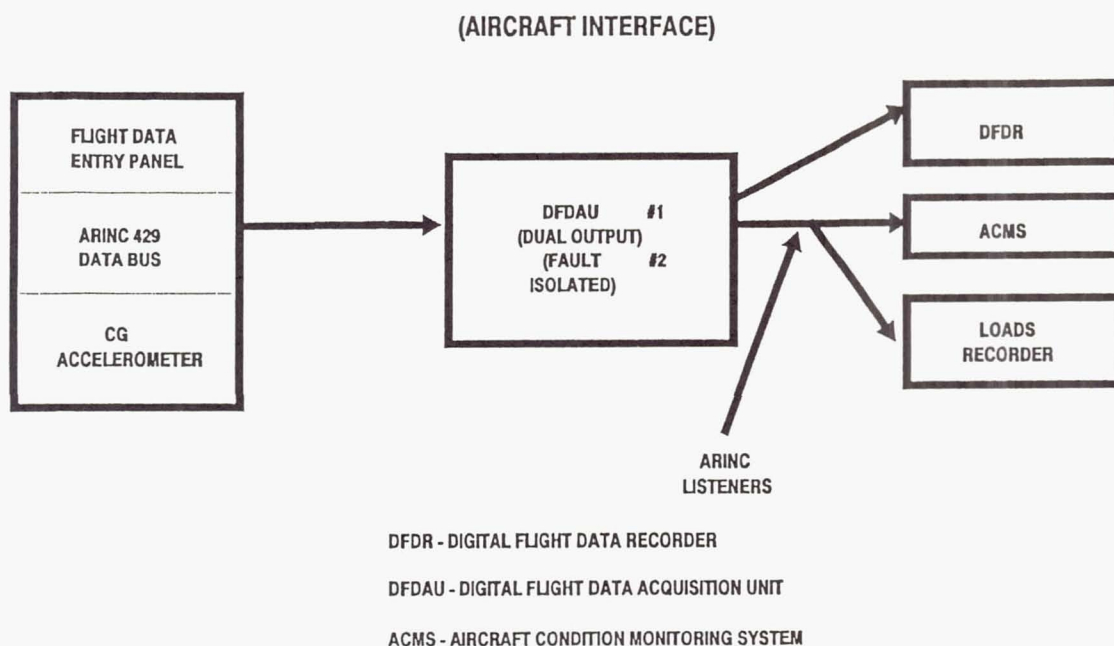


Figure 7  
**FAA Flight Loads Data Collection Program**

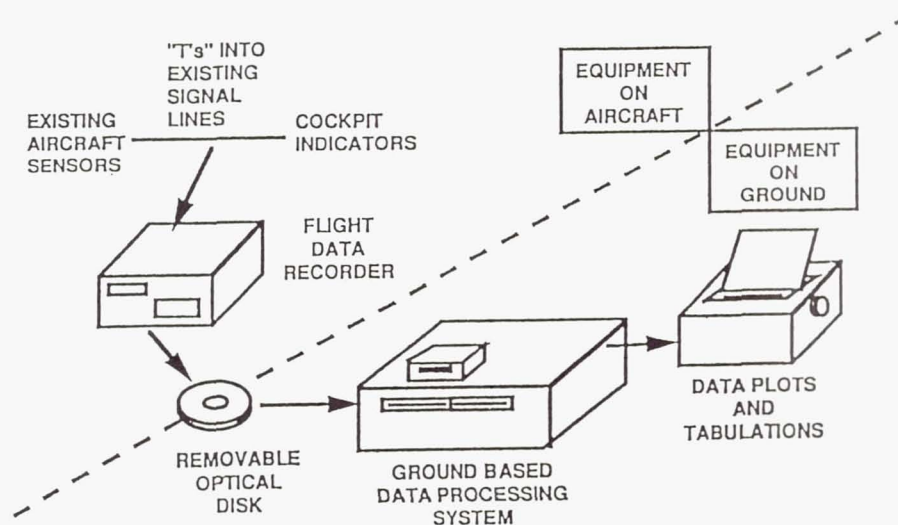


Figure 8  
**NEW RECORDER SYSTEM CONCEPT**



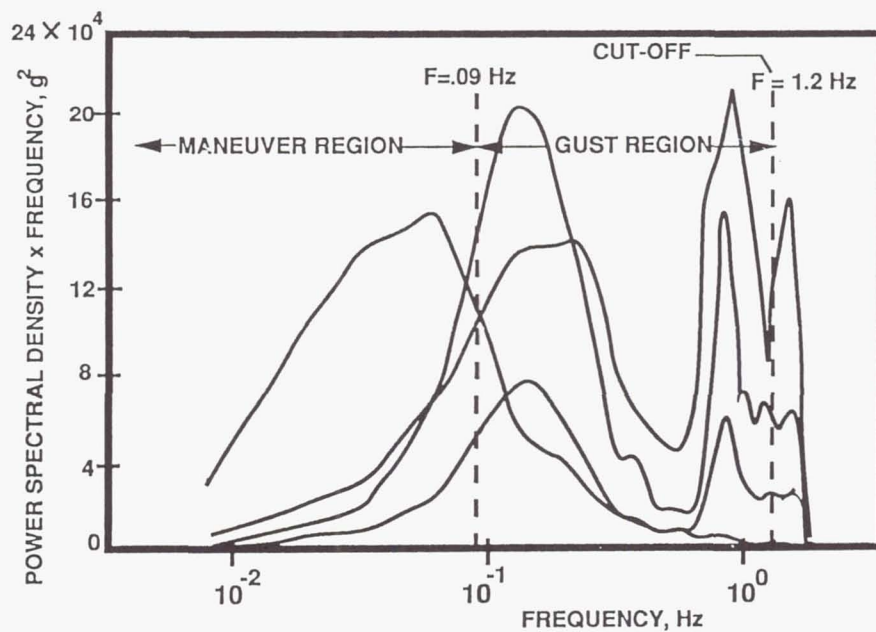


Figure 9

# SAMPLE L1011 NORMAL ACCELERATION POWER SPECTRA

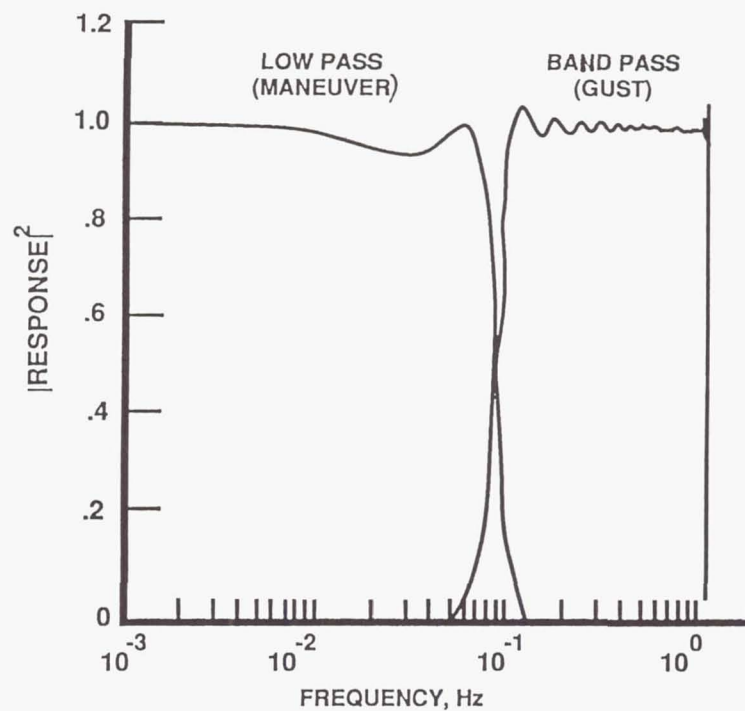


Figure 10

FREQUENCY RESPONSE OF NUMERICAL FILTERS USED TO SEPARATE GUST AND MANEUVER ACCELERATIONS, AND TO ELIMINATE ELASTIC RESPONSES ABOVE 1.2Hz

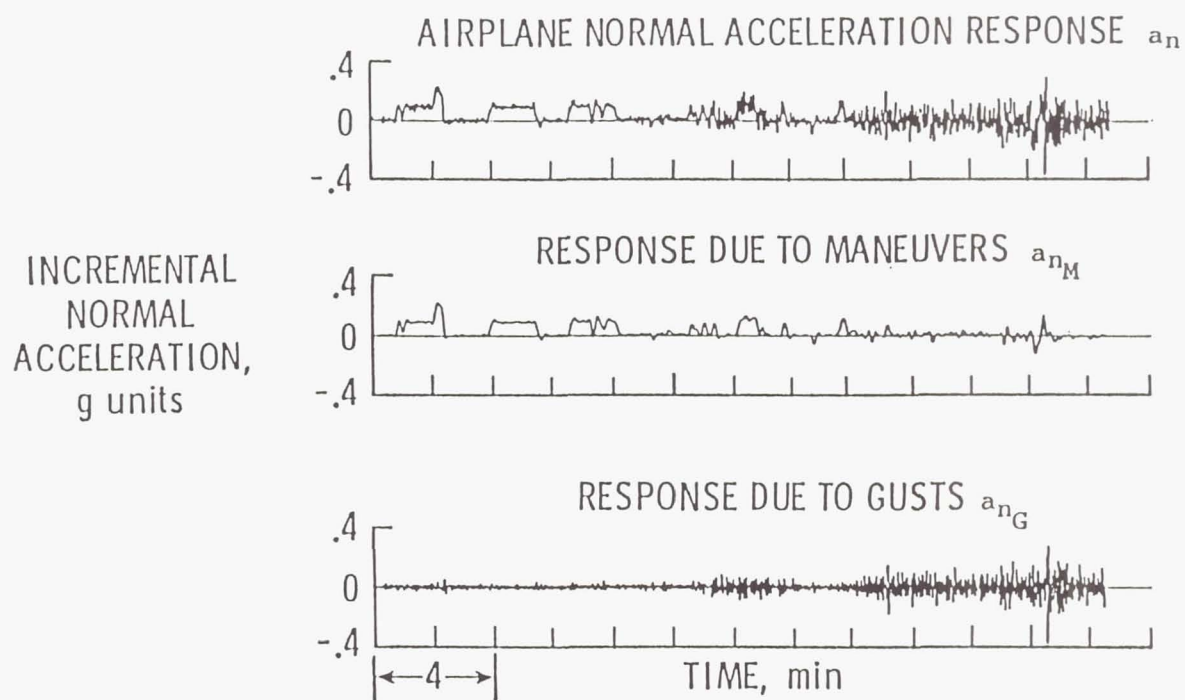


Figure 11  
FILTER SEPARATION OF NORMAL ACCELERATION TIME HISTORY

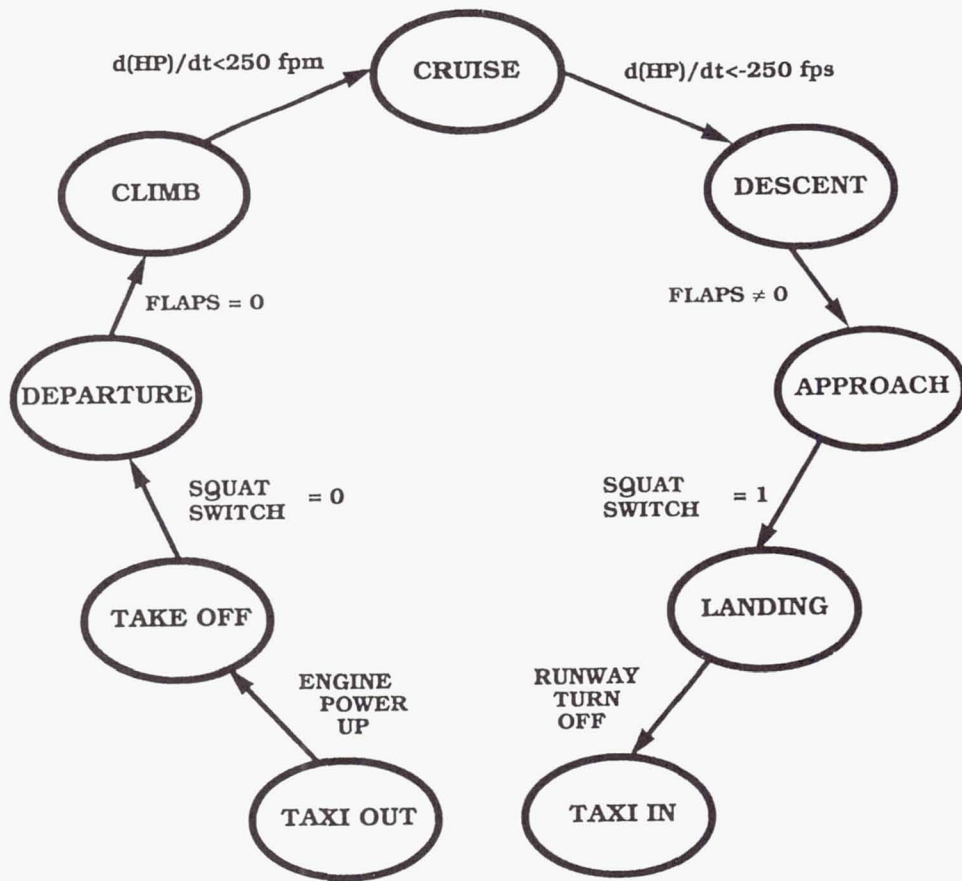


Figure 12  
FLIGHT PROFILE DETERMINATION

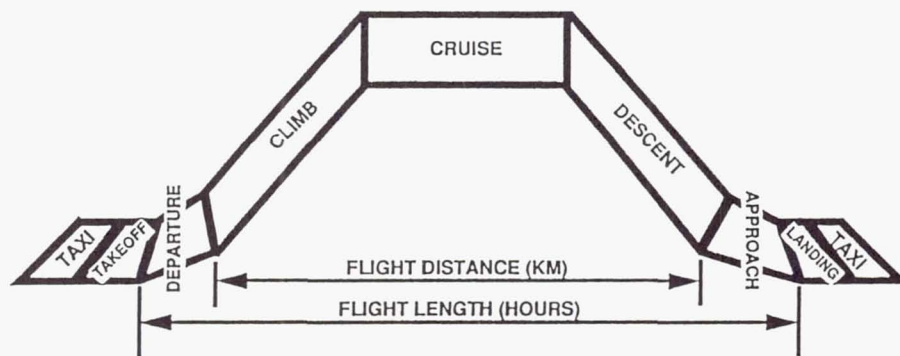


Figure 13  
FLIGHT PROFILE SEGMENTS



Figure 14

## OUTLINE OF PROPOSED ANALYSES

PHASE OF FLIGHT	ANALYSIS CATEGORY												
	FLIGHT PROFILE STATISTICS	ACCELERATION DEFINES QUANTITIES		CONTROL PARAMETERS					OTHER PARAMETERS				
	ALTITUDE, AIRSPEEDS, WEIGHTS, AND DURATIONS	NORMAL ACCEL.	LATERAL ACCEL.	RUDDER	AILERON	ELEVATOR	SPOILER	STABILIZER	FLAPS	POWER	PITCH	ROLL	YAW
PER FLIGHT SUMMARY													
TAXI OUT													
TAKE OFF													
DEPARTURE													
CLIMB													
CRUISE													
DESCENT													
APPROACH													
LANDING													
TAXI IN													

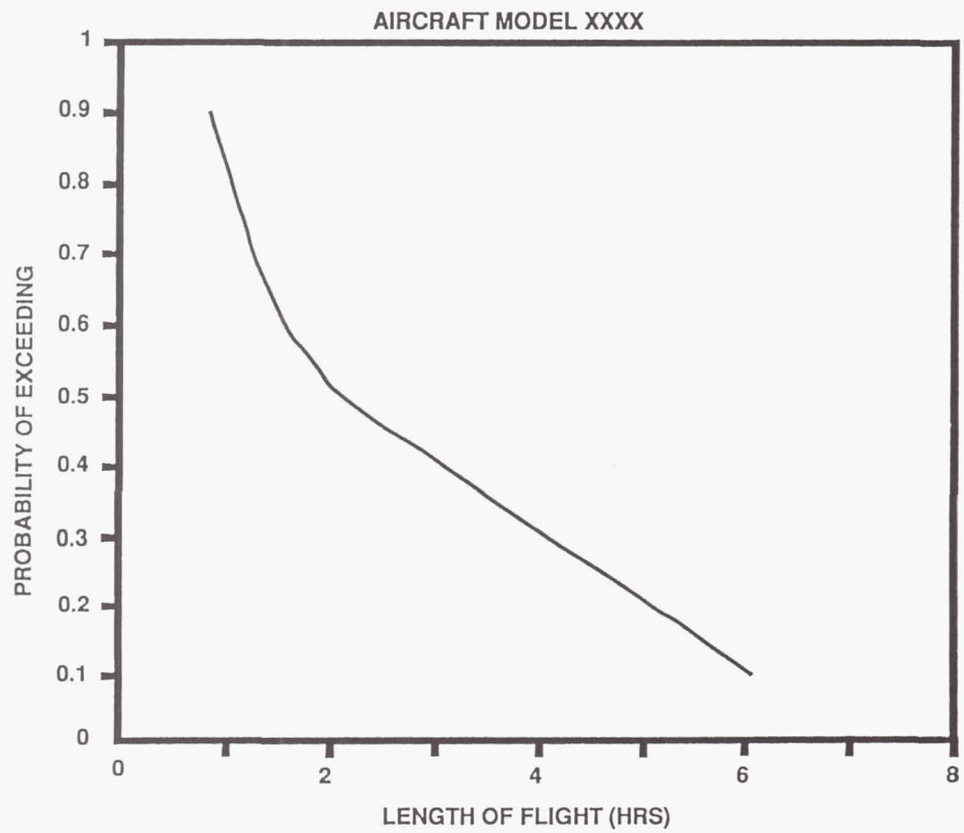


Figure 15  
DISTRIBUTION OF FLIGHT LENGTH

Figure 16  
CUMULATIVE FREQUENCY

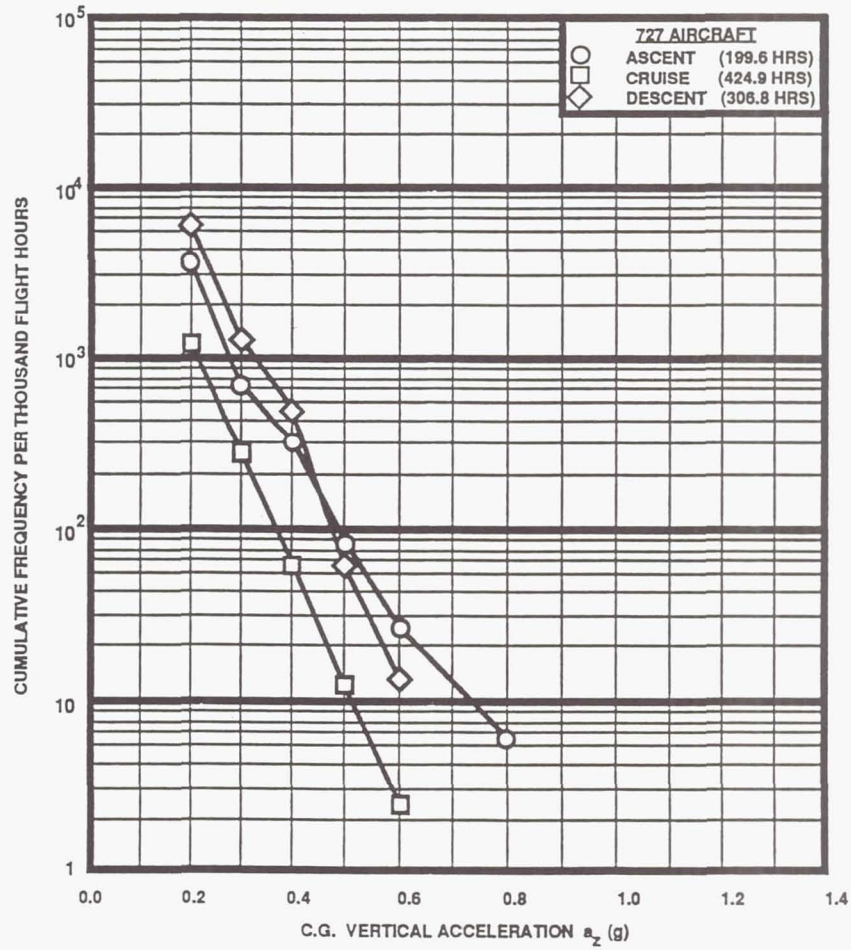




Figure 17

FLIGHT PHASE #: NORMAL ACCELERATION:  
PEAK-VALLEY COUNTS/HOUR

		PEAK NORMAL ACCELERATION (g UNITS)																					
		> 2.0	1.8 to 2.0	1.6 TO 1.8	1.4 TO 1.6	1.2 TO 1.4	1.0 TO 1.2	.8 TO 1.0	.6 TO .8	.5 TO .6	.4 TO .5	.3 TO .4	-.3 TO -.4	-.4 TO -.5	-.5 TO -.6	-.6 TO -.8	-.8 TO -1.0	-.9 TO -1.2	-1.2 TO -1.4	-1.4 TO -1.6	-1.6 TO -1.8	-1.8 to -2.0	< -2.0
FOLLOWING VALLEY NORMAL ACCELERATION (g UNITS)	> 2.0																						
	1.8 to 2.0																						
	1.6 TO 1.8																						
	1.4 TO 1.6																						
	1.2 TO 1.4																						
	1.0 TO 1.2																						
	.8 TO 1.0																						
	.6 TO .8																						
	.5 TO .6																						
	.4 TO .5																						
	.3 TO .4																						
	-.3 TO -.4																						
	-.4 TO -.5																						
	-.5 TO -.6																						
	-.6 TO -.8																						
	-.8 TO -1.0																						
	-1.0 TO -1.2																						
	-1.2 TO -1.4																						
	-1.4 TO -1.6																						
	-1.6 TO -1.8																						
	-1.8 to -2.0																						
	< -2.0																						

FOR ALTITUDE <4500 FEET

TOTAL FLIGHTS \_\_\_\_\_  
 TOTAL HOURS \_\_\_\_\_  
 TOTAL MILES \_\_\_\_\_  
 CALENDAR START/STOP \_\_\_\_\_

Figure 18

DEPARTURE: NORMAL GUST VELOCITY EXCEEDANCES:  
PEAK-VALLEY COUNTS/HOUR

PEAK  $U_{de}$ , FEET PER SECOND

	< -110	-90 to -110	-70 to -90	-60 to -70	-50 to -60	-40 to -50	-30 to -40	-25 to -30	-20 to -25	-15 to -20	-10 to -15	10 to 15	15 to 20	20 to 25	25 to 30	30 to 35	35 to 40	40 to 50	50 to 60	60 to 70	70 to 90	90 to 110	> 110
FOLLOWING VALLEY $U_{de}$ , FEET PER SECOND																							
> 110																							
90 to 110																							
70 to 90																							
60 to 70																							
50 to 60																							
40 to 50																							
30 to 40																							
25 to 30																							
20 to 25																							
15 to 20																							
10 to 15																							
-10 to -15																							
-15 to -20																							
-20 to -25																							
-25 to -30																							
-30 to -35																							
-35 to -40																							
-40 to -50																							
-50 to -60																							
-60 to -70																							
-70 to -90																							
-90 to -110																							
< -110																							

$U_{de}$  FOR ALTITUDE <4500 FEET

TOTAL FLIGHTS \_\_\_\_\_  
 TOTAL HOURS \_\_\_\_\_  
 TOTAL MILES \_\_\_\_\_  
 CALENDAR START/STOP \_\_\_\_\_

**AILERON SUMMATION**

BELOW		DYNAMIC PRESSURE PSF																			
		19	34	53	77	101	139	176	219	267	321	380	445	516	593						
		AIR SPEED																TOTAL			
BELOW		75	100	125	150	175	200	223	250	275	300	325	350	375	400						
DELTA AILERON LESS																					
-60																					
-50																					
-40																					
-30																					
-20																					
-10																					
0																					
10																					
20																					
30																					
40																					
50																					
60																					

Figure 19A - 727 DATA

**RUDDER SUMMATION**

		DYNAMIC PRESSURE PSF																	
BELOW		19	34	53	77	101	139	176	219	267	321	380	445	516	593				
		AIR SPEED KTS																	
BELOW		75	100	125	150	175	200	223	250	275	300	325	350	375	400	TOTAL			
RUDDER POSITION																			
-60																			
-50																			
-40																4			
-30																18			
-20		7	100	77	4	1		1	1		1			192					
-10		159	563	177	78	121	197	104	102	154	127	89	146	22	2039				
0		351	729	275	116	244	443	262	208	272	317	309	318	58	3904				
10		9	91	46	2											144			
20		10		2											12				
30		1													1				
40		1															1		
50																			
60																			

Figure 19B - 727 DATA

**STABILIZER SUMMATION**

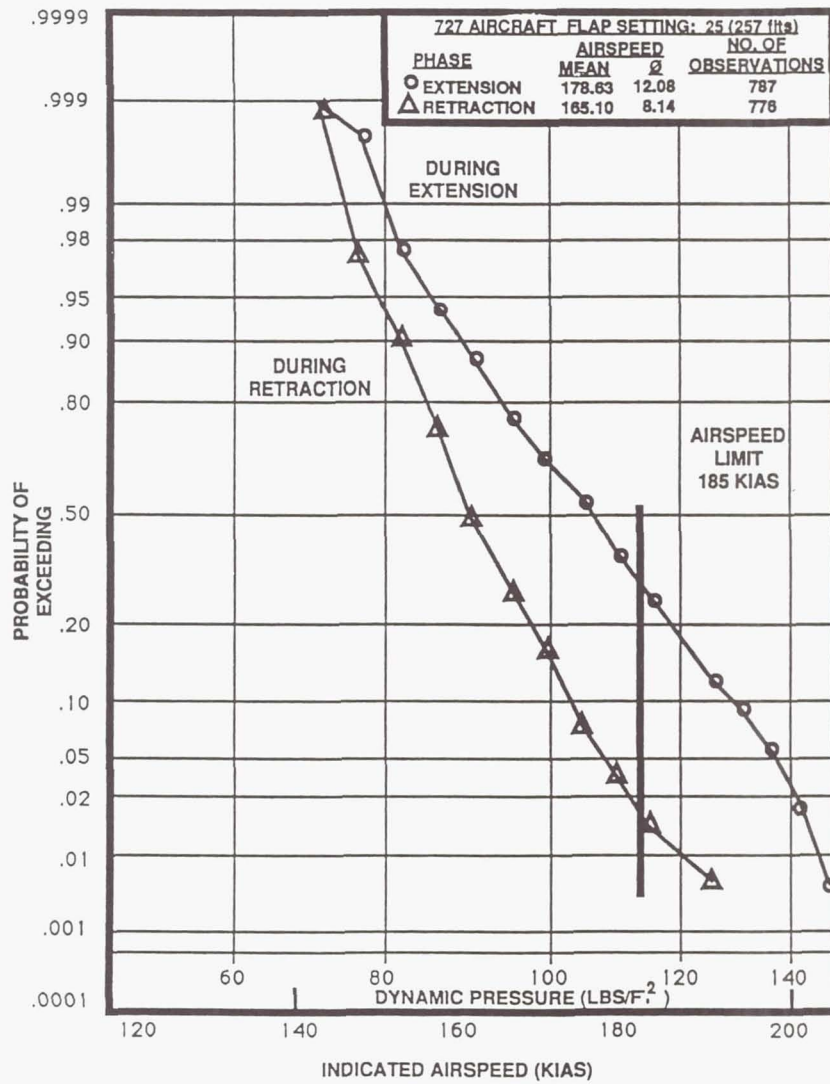
		DYNAMIC PRESSURE PSF																									
BELOW		19	34	53	77	101	139	176	219	267	321	380	445	516	593												
		AIR SPEED KTS																									
BELOW		75	100	125	150	175	200	223	250	275	300	325	350	375	400	TOTAL											
STABILIZER POSITION																											
	0																										
	10															1	1	2	23	109	235	413	117	901			
	20															2	60	180	165	270	417	374	292	203	46	2012	
	30															3	58	343	379	113	18	9	1				924
	40																									519	
	50																									2254	
	60																									181	
	70																										
	80																										
	90																										

Figure 19C - 727 DATA



Figure 20

# PROBABILITY OF EXCEEDING DURING EXTENSION AND RETRACTION



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# DAMAGE TOLERANCE FOR COMMUTER AIRCRAFT

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## SUMMARY

The damage tolerance experience in the United States Air Force with military aircraft and in the commercial world with large transport category aircraft indicates that a similar success could be achieved in commuter aircraft. The damage tolerance process is described for the purpose of defining the approach that could be used for these aircraft to ensure structural integrity. Results of some of the damage tolerance assessments for this class of aircraft are examined to illustrate the benefits derived from this approach. Recommendations are given for future damage tolerance assessment of existing commuter aircraft and on the incorporation of damage tolerance capability in new designs.

## INTRODUCTION

The Aircraft Structural Integrity Program (ASIP) was initiated in 1958 by the United States Air Force in response to fatigue failures which occurred in service aircraft in the fifties (ref. 1). The original version of the ASIP recognized the need for a fatigue analysis and test program for fatigue life validation. The fatigue methodology adopted was a reliability based approach commonly referred to as the "safe life" method. This approach was used in the development of USAF aircraft in the sixties. It was found that this approach did not adequately address the threat of manufacturing and service induced damage. The safe life approach did not preclude the use of "brittle" materials at high stresses, and therefore, the structures were not tolerant to defects in the structure derived from the manufacturing process or from in-service maintenance of the aircraft. Consequently, aircraft designed and qualified by this approach suffered premature failures when operated in the service environment. There were many aircraft in this category, such as the F-5, KC-135, B-52 and F-111. Each of these aircraft suffered failures in service at times considerably less than the laboratory demonstrated safe life. Service failures resulted in costly redesign and modification programs on many aircraft. Further, the safe life approach did not include a means for establishing an inspection program. There was no rational process for determining where, when, how and how often inspections should be accomplished to maintain operational safety. These deficiencies in the safe life approach motivated the USAF to adopt the damage tolerance approach. After this approach was applied to the design of the B-1 bomber and to the modification program of the C-5 and F-4 aircraft, it was formally established in MIL-STD-1530A released in December of 1975. Since that time all of the major weapons in the USAF have had their Fleet Structural Maintenance Plans (i.e., the how, when and where to inspect or modify the aircraft to maintain safe and economical operations) updated by the damage tolerance approach.

There has been a somewhat different path to damage tolerance in the commercial world. The Martin 202 wing failure in 1948 and the deHavilland Comet fuselage failures in 1954 led to the adoption of the fail-safe design concept. It was concluded after the Avro 748 wing failure in 1976 and the Boeing 707 horizontal tail failure in 1977, where both aircraft had been designed to be fail-safe, that fail-safety alone was not adequate to protect an aircraft from catastrophic failure. It was further concluded that an inspection program based on damage tolerance should be added to the maintenance program. In the United States, this was accomplished in 1978 with the release of Amendment 45 to FAR Part 25.



The success of the damage tolerance approach for military aircraft and for large transport category aircraft has firmly established this process for maintaining safety. The question is: "should the damage tolerance approach be extended to the small commuter aircraft that are in fare paying passenger service?" A partial answer to this question can be obtained from an examination of the history of the development of commuter class aircraft. Some of these aircraft have been certified under FAR Part 23 requirements and some have been certified under FAR Part 25 requirements. However, not all of the Part 23 aircraft or the Part 25 aircraft have been certified under the same rules. Improvements were made in the basic certification requirements as a result of operational aircraft experience. Examples of these changes for Part 23 aircraft include the following: in 1953 the rule was modified to include the pressure cabin; in 1969 the wing and wing carry thru were added and in 1989 the empennage was added. Consequently, there are aircraft operating in commuter service that have little, if any, assessment of their resistance to cracking by either the safe life or damage tolerance method.

The application of the safe life approach for commuter aircraft has the same difficulties as those experienced by the Air Force - namely from the use of brittle materials in locally high stress applications that may be subject to accidental damage in production or service. In addition, there is no mandatory requirement for a fatigue test of commuter aircraft. In those cases where testing was not accomplished, the safe life was determined based on the results of a linear cumulative damage analysis that was divided by a scatter factor. The scatter factor is a number greater than one that is intended to account for uncertainty in the analysis as well as the material and manufacturing quality variability. As indicated above, some of the older aircraft developed under FAR Part 23 have been certified without even a fatigue analysis. There are no life limits or a rationally based inspection program on these older aircraft, and in many cases, there is currently no basis for establishing a life limit or a rationally based inspection program.

However, there have been few commuter aircraft accidents attributed to structural failure. This is believed to be the result of the fact that the generic stresses in these aircraft are generally low. The strength requirements for maneuver and gust combined with the normal conservatisms that are used in stress analyses cause these stresses to be low. The situation may be different for local areas where the absence of detailed stress analysis or adequate testing could lead to failure. At the present time, the risk of this occurring is unquantified.

It is the purpose of this report to discuss the damage tolerance approach for commuter aircraft and how the damage tolerance process could be used to develop an inspection program or alternatively life limits. The findings of the available damage tolerance assessment efforts will be summarized and recommendations made for damage tolerance analyses of the structural components of both existing and new commuter aircraft.

## THE DAMAGE TOLERANCE APPROACH

The approach that has been used for damage tolerance assessments in the Air Force has not significantly changed over the last eighteen years. There are, however, differences in the emphasis that has been placed on the individual tasks among the aircraft that have been assessed for damage tolerance capability. This is a natural consequence of the differences among the aircraft and the differences in the data base available at the start of the assessment. As would be expected, many of the tasks that are performed for the damage tolerance assessment are precisely the same as that accomplished for the classical fatigue analysis which is the basis for establishing design stresses for aircraft that use the safe



life method. The timing of the damage tolerance assessments that have been performed under sponsorship by the Air Force is shown in figure 1. As seen in this figure, both large and small aircraft have been subjected to an assessment. The effort required for an assessment generally increases as the size of the aircraft increases. Also, it depends heavily on the data base available at the time of the assessment and the criticality of the structure to the usage environment. For example, the assessment on the F-4 required more effort than normal because the loads data had not been verified by a flight loads survey. A flight loads survey was conducted on this aircraft to provide this needed data. Also, the assessment performed on the T-38 for operations in the severe usage associated with lead-in-fighter training took considerably longer than the assessment of the F-15 in a usage for which it was designed even though the structure of the F-15 is more complex than the T-38. The damage tolerance process and the details of many of the assessments performed by the Air Force are discussed in reference 2. As discussed in this reference the process was divided into the following six tasks:

- Identification of critical areas
- Development of the stress spectrum
- Establishment of the initial flaw criterion
- Establishment of the operational limits
- Development of the Force Structural Maintenance Plan
- Development of a fracture mechanics based tracking program

The first five of these tasks are illustrated in figure 2.

It was found for all of the damage tolerance assessments that the identification of critical areas (i.e., those areas of the structure where an inspection or modification is expected to take place during the life of the structure) is augmented from a knowledge of the service problems, laboratory fatigue testing, stress analyses and strain surveys from ground and flight tests. For many of the older commuter aircraft there have been service cracking problems. The manufacturer and the FAA have addressed these areas through airworthiness directives. These areas would be a starting point for the analyst to locate potential adjacent area cracking points. The fatigue test data base, however, for most of the commuter aircraft is nonexistent. Also, in most cases there is little if any data on the stress distributions on the structural components. Therefore, the identification of critical areas of the structure will need to be determined from analytically derived stresses from areas of the structure where there are either stress concentrations or materials used that have poor fracture properties. The lack of a fatigue test or a strain survey data base for a complex military aircraft structure would be a significant detriment to the successful completion of an assessment. However, the inherent simplicity of most commuter aircraft structures makes this task considerably easier. Further, for many of the commuter aircraft, there has been a classical fatigue analysis performed. These fatigue analyses required that the critical areas of the structure for fatigue be located. The areas of the structure that are critical for safe life are also critical for damage tolerance. There could, however, be some areas that are critical for damage tolerance that are not critical for fatigue. Therefore, an existing fatigue analysis is extremely helpful, but not always sufficient for the damage tolerance analysis. Therefore, the analyst needs to initially identify the critical areas in somewhat conservative fashion and then reduce the number for a detailed examination by a preliminary



screening process. This preliminary screening activity may involve some finite element modeling followed by a simplified evaluation of the stress spectra and a fracture analysis.

The stress spectrum development was typically the most difficult to perform of all the tasks for the damage tolerance assessments. Much of the difficulty arises because of the need for accuracy. Accuracy of the stress spectrum is important for the commuter aircraft whether the analysis is for safe life or for damage tolerance. As stated in reference 2, there are the following three phases in the development of the stress spectrum for an aircraft:

- Determination of the aircraft usage
- Determination of the external loads
- Determination of the stresses

With some exceptions, there is considerable commonality in the usage of commuter aircraft. The maneuvering load factors are, in general, similar to the large transport category aircraft. Also, the maneuver loads, when compared to low level turbulence, are typically benign. The major usage differences that are significant are number of hours per flight and altitude flown. The number of hours per flight determines the number of ground-air-ground cycles and the altitude flown determines the turbulence spectrum. It is believed that the atmospheric turbulence has been characterized well enough to provide an adequate spectrum for these aircraft.

The second phase in the development of the stress spectrum for the airframe is to determine the external loads on the structure for a given maneuver or gust condition. This is a problem for many commuter aircraft because of the fact there has been little, if any, effort expended to perform a flight loads survey to determine the loads empirically. This may not be a problem for the wing since the wing configurations on commuter aircraft generally lend themselves to analysis. However, this problem may be particularly significant for the empennage and consequently on the aft fuselage. The interpretation of the results of the final analyses for obtaining operational limits should be done with a recognition of the possible shortcomings of this phase of the stress spectrum development.

The third phase of the stress spectra development is to determine the detail stresses in the critical areas resulting from the external loads. It was found in all of the military aircraft damage tolerance assessments that knowledge of the detail stresses was lacking. This was due primarily to the lack of computer capability when these aircraft were designed. The qualification process substituted laboratory testing for detailed analyses. It is essential for either a safe life or a damage tolerance analysis that there be an adequate understanding of the stresses in the structure. With the computational capability that exists today, this is not a significant inhibitor to the process. Further, the simplicity of the commuter aircraft structures makes these analyses economically viable.

The desired end product of the stress spectra development is a flight-by-flight spectrum of stresses for each of the critical areas. The damage tolerance assessments that have been performed have shown that the ordering of the flights in the spectrum is not significant if the ordering is random. This has also been found to be true for the commuter aircraft stress spectra. In addition to the development of the baseline spectra which are intended to represent average usage, it is desirable to generate variations of these spectra that represent the expected range of usage of the vehicle. The successful completion of the fracture analyses and tests with these spectra provide confidence that there is a sound data base for use in the development of a maintenance plan for the aircraft.



The third task of the damage tolerance assessment is to determine the initial flaw size to be used for the subsequent fracture analysis. The data derived from operational experience generated over the last eighteen years has shown that for fastener holes in airframe structure, an initial corner flaw of 1.27 millimeter radius is adequate to provide safety of flight. There is no known case where an aircraft has failed in less than the time required for this size defect to grow to critical size. It has also been found that for airframe structure that a semicircular surface flaw with a radius of 3.175 millimeter is similarly adequate. No studies have been accomplished to show that the size of the aircraft has an influence on the probability distribution of flaws. A large aircraft would logically have more opportunities per aircraft for a rogue defect. It is not clear, however, how to adjust the size of the rogue defect for the smaller aircraft, and consequently this has not been done. Therefore, the initial flaw sizes (i.e., at the time the aircraft is manufactured) above are suggested for a damage tolerance assessment of a commuter aircraft. Where needed, a continuing damage size of 0.127 millimeter corner flaw in a hole should be used. It is further recommended that the in-service inspection flaw size be based on a probability of detection of 0.9 with a 95 percent confidence. In-service inspections have been a problem for commuter type aircraft since the safe life certification process places no emphasis on inspectability, and consequently there are many details that are difficult to inspect. Some of these problems have been diminished because of the recent developments in low frequency eddy current that will permit reliable inspections of structure that would have previously been considered virtually noninspectable. This technique is commercially available and performs inspections without the removal of the fasteners. It is especially suitable for many locations in commuter aircraft because the depth of penetration required for second and third layer inspections is relatively small compared to its capability.

The operational limits for commuter aircraft are derived from the fracture analyses and tests based on the stress spectrum for each critical area in combination with the initial flaw to be used for that area. As reported in reference 2, the predominant problem with aircraft is fastener holes. Thus, there is no problem in obtaining adequate stress intensity solutions for the critical areas of the structure. The test verification of the analytical flaw growth is normally an essential part of the process to obtain the spectrum interaction effects which may be significant in the determination of the operational limits. This is initially accomplished at the coupon level where the test specimens have the same material, thickness, hole geometry and, if necessary, fastener load transfer. The full-scale fatigue tests are used, when possible, for final validation of the crack growth analysis.

The Fleet Structural Maintenance Plan (FSMP), as indicated previously, is the plan that describes the inspection and modification program for the commuter aircraft during its anticipated operational life. The damage tolerance process is shown graphically in figure 3. The inspection program is determined from the calculation of the operational or safety limits. The initial safety limit for a fastener hole, for example, is the unfactored crack growth life from an initial 1.27 millimeter crack to critical crack length at limit load. It is recommended that the safety limit be divided by a factor of two to determine the inspection threshold. For a structure that is noninspectable or is not planned to be inspected, the safety limit could be used as a life limit. This should be done with care on those structures where there is reason to question the analysis because there has not been a flight loads survey to validate the loading. In most of the damage tolerance assessments performed on military aircraft there were many parts found that were extremely critical. In many cases these critical areas were revealed through in-service failures and through full-scale fatigue tests. As indicated in reference 2, these problems were addressed individually and an approach was determined that preserved safety at minimum cost.



## DAMAGE TOLERANCE EXPERIENCE WITH OLDER AIRCRAFT

### B-52 Companion Trainer

In 1980, the USAF was interested in procuring a B-52 companion trainer aircraft. This aircraft would be used to reduce the training requirements in B-52 aircraft and therefore would be required to fly its missions. These missions included a considerable amount of severe low altitude usage. The candidates for this aircraft included aircraft whose structural integrity was unknown to the Air Force. Previous success with damage tolerance on other aircraft motivated them to make an initial damage tolerance assessment of these aircraft to see how well they performed against this criterion. The following aircraft were examined:

- British Aerospace 125 Series 700
- Lockheed S-3A
- Israeli Aircraft Industries Westwind
- Dassault Mystère 10
- Dassault Mystère 20
- Gates Learjet 35
- Rockwell Sabliner 65

The contractors were asked to provide basic information on the material used and the geometry of critical details in the wing, fuselage and empennage as well as one g stress and stress per g. These aircraft were assessed against an executive mission which was representative of commuter aircraft operations and also against the severe B-52 mission. The maneuver and gust exceedance functions for these missions were converted to a flight-by-flight spectra for the subsequent fracture analysis. The findings of this assessment were that there was considerable scatter in the material selection and in the capabilities of these aircraft. Three of the candidates were found to have a damage tolerance derived safety limit of over 100,000 hours for the executive mission and these same three had an adequate safety limit for use in the B-52 mission. One of them had a small safety limit in the executive mission and three of the candidates had a less than desirable safety limit in the B-52 mission. One of these aircraft was unacceptable for the B-52 mission. This assessment showed that a modest effort could provide a significantly better basis for screening candidate aircraft for a given usage. It also demonstrated that the generic stress levels chosen for commuter aircraft provided a good damage tolerance capability. The reason that some aircraft fell short of the desired goal was because of the lack of damage tolerance capability in some of the design details.

### T-37

As a result of the anticipated cancellation of the T-46, the Air Force initiated a review in April of 1986 of the T-37 aircraft to determine its potential for a life extension. The T-37 is an aircraft used by the Air Force for initial pilot training of its air crews. Consequently, it is subjected to severe usage as indicated by the load factor exceedance function shown in figure 4. The development of the aircraft was initiated by the Air Force with a contract with Cessna in January of 1953. The material selection included 7075-T6 forgings and extrusions, 7075-T73 extrusions, and 2024-T42 sheet. The first aircraft was delivered in 1955 with a gross weight of 6600 pounds. The initial fatigue life investigations were based on a required service life of 8,000 hours and 20,000 landings. In 1968, a front wing spar failed on a T-37 with approximately 6,000 flight hours, and another aircraft suffered a failure in the horizontal tail.

In addition, there have been numerous cases of stress corrosion cracking of the 7075-T6 forgings. As a result of structural modifications and a full-scale fatigue test, the life of the aircraft was extended to 18,000 hours based on the safe life approach. By April of 1986, the weight of the 644 remaining aircraft had been increased to 7200 pounds. The high time aircraft had 15,322 hours and the average aircraft had 11,748 hours. The oldest aircraft was approximately 28 years old and the average aircraft was approximately 23 years old.

The Air Force felt that there was not an adequate understanding of the structure of this aircraft from which to determine what modifications should be incorporated. Also, the inspection program for the critical details and the life remaining for the structure were not well established. To obtain a solution to these problems, the Air Force contracted with Cessna for a damage tolerance assessment. Although this aircraft was designed for acrobatic maneuvers that are outside the capability of the typical commuter aircraft, it was a small aircraft with construction that was similar to that used in commuter aircraft. Therefore, the damage tolerance assessment for this aircraft should encounter similar technical challenges as the damage tolerance assessment of any commuter aircraft. For this aircraft, the data base for the damage tolerance assessment was augmented by a flight loads survey to validate the loads on the wing and the empennage. There were forty-nine candidate critical areas identified by a screening process. Preliminary analyses were able to remove all but eighteen of the areas from consideration. Detail analyses were performed on these eighteen locations. Many of the critical locations in the aircraft were in the wing carry thru and in the fuselage. These areas had been subjected to a number of modification programs over the years, some of which involved the addition of steel straps. Some of these structural details were found to be extremely critical. The results of the T-37 assessment showed that the safety limits of the structure ranged from less than 1000 hours to over 60,000 hours. The safety limits of some of the details are shown in figure 5. It was found that all of these aircraft were operating with flight hours in excess of the safety limits. By policy, the Air Force would have grounded these aircraft until an inspection had been performed. Since some of these inspections required considerable downtime and continued operation of these aircraft was essential for meeting training requirements, the aircraft remained in service based on a risk assessment. The inspections were performed on an accelerated basis. The damage tolerance analysis effort clearly identified those areas that needed modification in order to extend the life of this aircraft.

#### Variable Stability Learjet

The test pilots school at Edwards Air Force Base in California uses an aircraft with a capability for variable stability as a part of its training program. The B-26 aircraft that had been used for this activity failed catastrophically from a crack in the wing structure in March of 1981. Consequently, the school procured a Gates Learjet model 24D to be modified to perform this function. To preclude a possible reoccurrence of a structural failure, the school contracted with Gates to perform a damage tolerance assessment of this aircraft. The Learjet was certified by the FAA under Part 25 of the Federal Air Regulations. Although fail safety was incorporated in the design, the damage tolerance analysis was performed on the basis of slow crack growth.

Since the usage was known to be more severe than the design usage for this aircraft, a program was initiated to collect data from actual operations of the aircraft. This was accomplished with an eight channel recorder that recorded both accelerations and strains for 334 flights (560 hours). The comparison of the normal acceleration for design and the variable stability usage is shown in figure 6.



Based on the recorded data, a flight-by-flight spectrum was developed for the damage tolerance analysis. The analysis included the wing, empennage and engine mounts. The wing analysis revealed that the initial safety limits for all of the points analyzed exceeded 100,000 flight hours. In addition, linear cumulative damage was used to calculate the mean lives of the wing critical locations. It was found that in most cases these mean lives were smaller than the safety limits. However, they were also approximately 100,000 hours. The contractor recommended that at 3,600 hours the wing be removed from the aircraft and subjected to visual, eddy current and X-ray inspections. The motivation for this inspection was apparently driven by a recommendation in the basic maintenance program for the aircraft. It was not derived from rational damage tolerance considerations.

The engine mount on the Learjet was a welded steel part manufactured from 4130 heat treated to 1,240-1,450 MPa. The linear cumulative damage analysis for the engine mount indicated that the part had a mean life of 764,000 hours. The damage tolerance analysis from a 1.5 millimeter corner flaw in a lightening hole revealed that the safety limit was less than 2,500 hours. This significant difference between the safe life and damage tolerance analysis is typical of materials that have an inherent sensitivity to flaws. The F-111 experience has shown that steel parts are subject to manufacturing damage as well as in-service damage. Unanticipated failures in cold proof tests have demonstrated this fact.

#### Piper PA-42

Another damage tolerance analysis of a commuter aircraft structural detail was conducted in 1990. This was a wing lower surface aluminum detail from the Piper (PA-42) Cheyenne IIIA. This detail was chosen because Piper had developed the data base from which this analysis could be easily accomplished. Piper had created a block spectrum of stresses for 15,000 hours of 1,000 hour blocks. This data was used to generate a flight-by-flight spectrum from which a damage tolerance assessment could be accomplished. The comparison of the stress exceedance function for the PA-42 usage and the C-5 usage is shown in figure 7. The similarity of these functions is remarkable.

It was found that stresses in this generic location in the wing were quite low and safety limit from a fastener hole in this aluminum structure was 52,500 hours. The crack growth function is shown in figure 8. This result was compared with results obtained with safe life methodology. First, the spectrum was analyzed by linear cumulative damage based on the S-N data in MIL-HDBK-5E. It was found that the unfactored life was 220,000 hours for a  $K_t$  of four in 2024-T3 aluminum. For a  $K_t$  of five for the same material, the unfactored life was found to be 26,400 hours. The decision on the selection of the effective  $K_t$  normally must be made from test data. It cannot be made on the basis of the geometric stress concentration only. This example illustrates the uncertainty that exists when test data is not available. The spectrum was also assessed based on the S-N curve from reference 3. It was found that the unfactored life was 29,200 hours. If a scatter factor of eight was used as recommended in reference 3, then the life would be 3,650 hours for this location. It is noted that the  $K_t$  of five data from MIL-HDBK-5E provides an unfactored life that is close to the unfactored life from reference 3. A study was performed to examine this location supposing that a 4340 steel strap had been used as a repair. The analysis of this steel part using an initial crack of 1.27 millimeters showed that the safety limit was 1,320 hours. The crack growth function for the steel is shown in figure 9. However, based on the S-N data in MIL-HDBK-5E for this same material at a  $K_t$  of 3.3, the unfactored life was found to be 121,000 hours and the factored life based on a scatter factor of eight was 15,125 hours. This illustrates the flaw

sensitivity and the lack of damage tolerance for the steel. It is seen that the safe life approach would not have rejected steel in this location for an aircraft life of 15,000 hours with no inspections.

### DAMAGE TOLERANCE ANALYSES OF OLD AIRCRAFT

A preliminary effort has been made to establish the urgency for a damage tolerance assessment of some of the older commuter aircraft. Although some aircraft in this category appear to have maintenance plans and retirement times that are adequate to protect their safety, there are others where the risk of failure is not known even in qualitative terms. The aircraft in this latter category should be subjected to damage tolerance assessments as soon as possible. However, all of the commuter aircraft should have their maintenance plans based on the damage tolerance approach. For structures that are noninspectable, the retirement time should be based on one half of the safety limit as defined above unless there has been a validation of the loads through a flight loads survey. In this case, the retirement time should be based on the safety limit.

### DAMAGE TOLERANCE ANALYSES OF NEW AIRCRAFT

The emphasis in the structural design of new commuter aircraft should be to eliminate, as far as practical, the need for in-service inspections to be performed to protect flight safety. However, it should be the intent that these aircraft be designed to be inspectable. There should be inspections to search for corrosion damage, in addition to inspections performed on a sampling basis to examine the structure for unexpected cracking.

It appears viable to design a commuter aircraft to an initial flaw concept and select the materials and establish the stresses such that no inspections are required. The stresses for the current population of commuter aircraft are such that if this were done there would be little, if any, weight impact of designing the structure to be free of inspections. The safety limit should be set at twice the desired service life of the aircraft. If life extension is required, then the inspection process would be initiated.

### CONCLUSIONS

The results shown above for commuter aircraft indicate that many of the parts of the structure are damage tolerant based on the current certification program. However, as seen in the T-37 assessment, the Gates Learjet assessment and the sensitivity study on the Piper PA-42, there could be details in the structure that could pass the safe life requirements and still be a potential safety hazard. The technology for evaluating existing commuter aircraft is well established. The capability exists to assess each of the aircraft in this category and make a determination on life limits or an inspection program to extend the life. The nondestructive inspection capability exists such that it is likely that an inspection program can be identified that will permit "retirement for cause" rather than retirement by flight hours.



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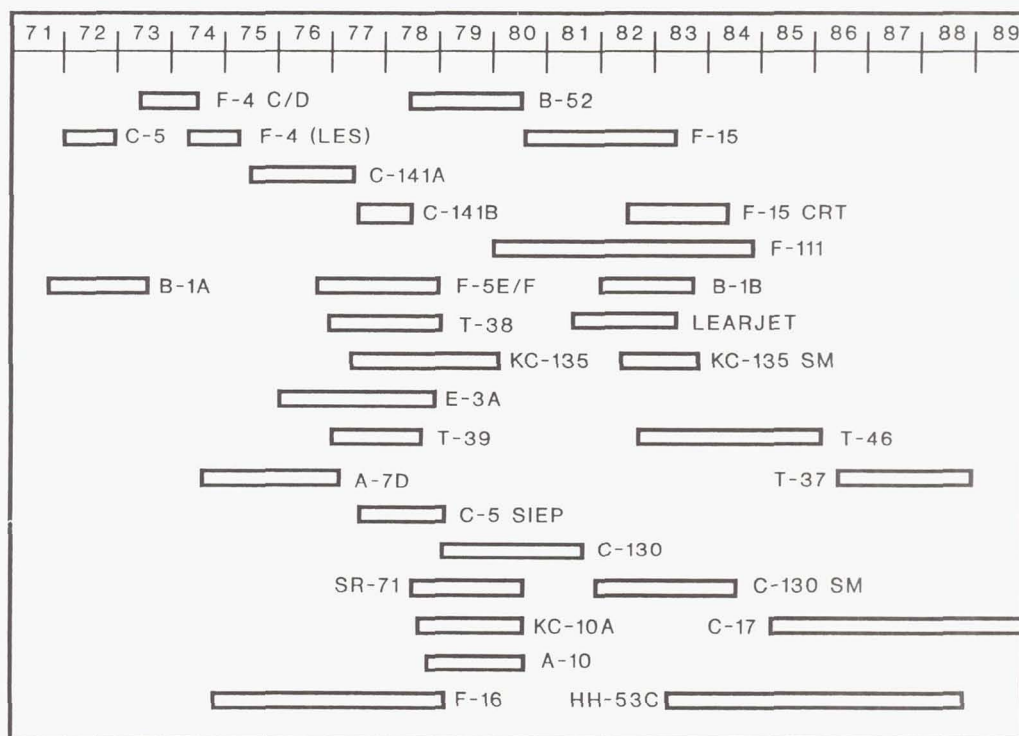


Figure 1. - Damage Tolerance Assessments of Aircraft

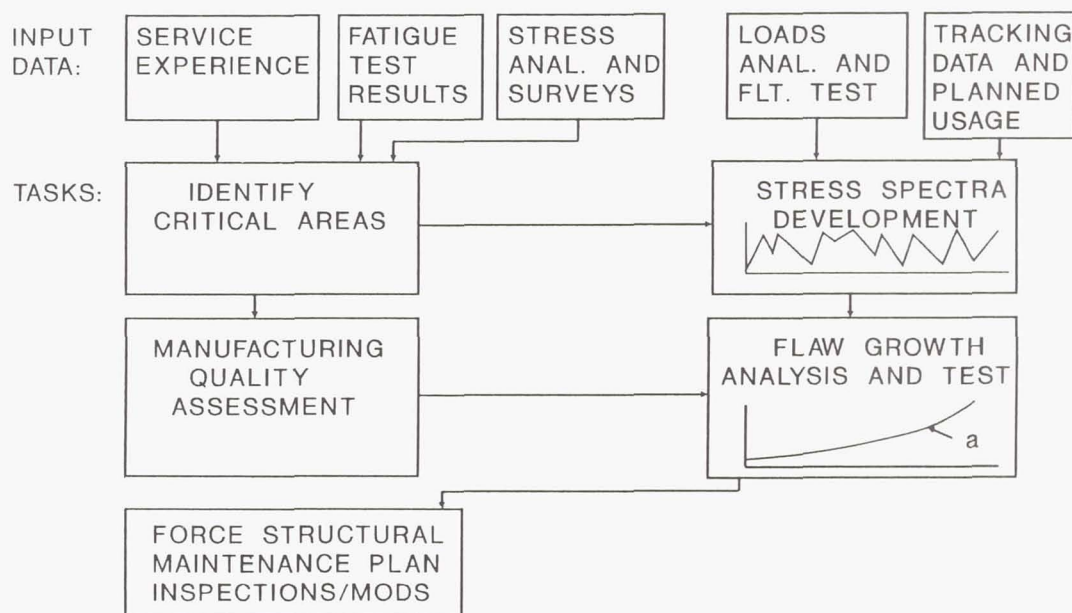
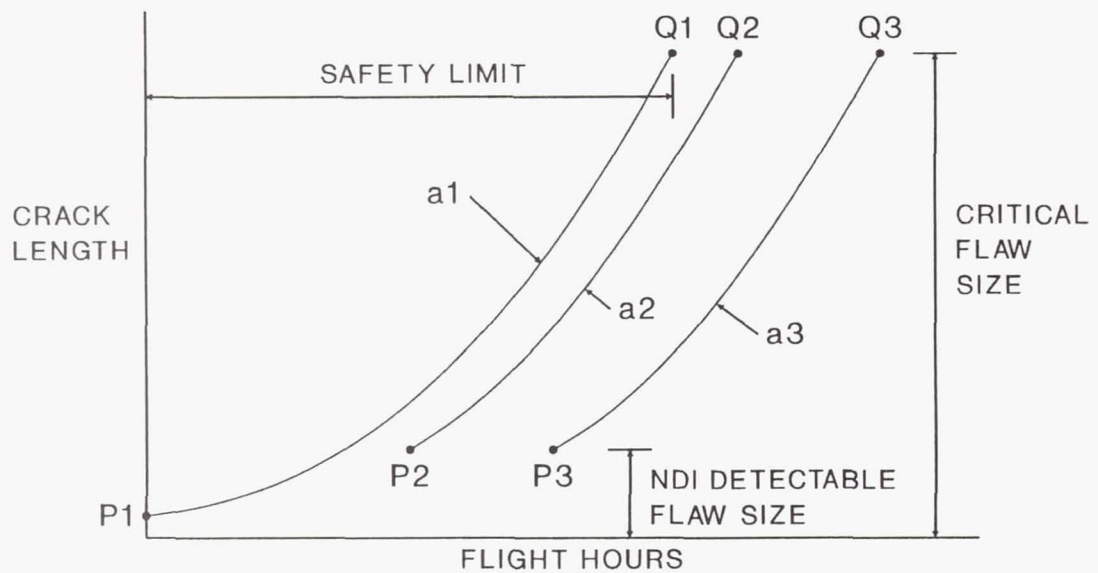


Figure 2. - Damage Tolerance Assessment Approach



FIRST INSPECTION AT 1/2 TIME TO CRITICAL FLAW SIZE  
 SECOND AND SUB. INSPECTIONS AT 1/2 TIME TO CRITICAL  
 FROM NDI FLAW SIZE PLUS TIME AT PREVIOUS INSPECTION

Figure 3. - Damage Tolerance Process - Slow Crack Growth

EXCEEDANCES PER 1000 HOURS

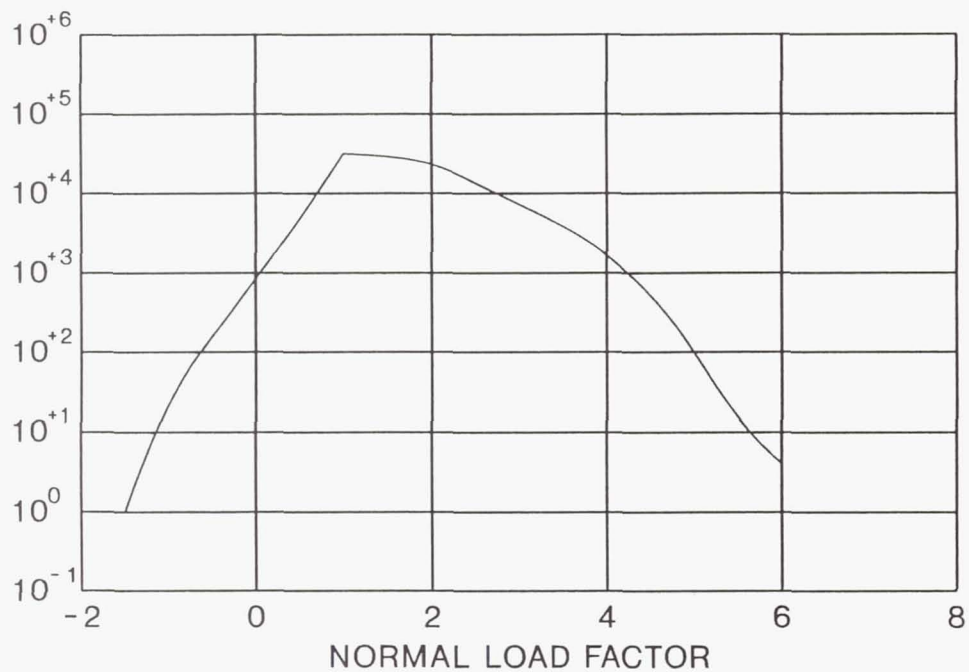


Figure 4. - Load Factor Exceedance for the T-37

LOCATION	MATERIAL	SAFETY LIMIT
WING CARRY THRU		
FORWARD UPPER CAP	2024-T42	>60,000
FORWARD LOWER CAP	2024-T42	9,822
FORWARD FORGING	7075-T6	1,930
FORGING LUG HOLE	7075-T6	874
WING FRONT SPAR		
LOWER CAP LUG	7075-T73	2,152
LOWER CAP FLANGE	7075-T73	32,150
LOWER CAP LUG HOLE	7075-T73	2,747
HORIZONTAL STABILIZER		
FRONT SPAR CAP	7075-T6	2,172
FUSELAGE CANOPY RAIL		
AFT LATCH CUT-OUT	2024-T42	>60,000
ATTACHMENT HOLE	2024-T3	>60,000

Figure 5. - Safety Limits for T-37 Details

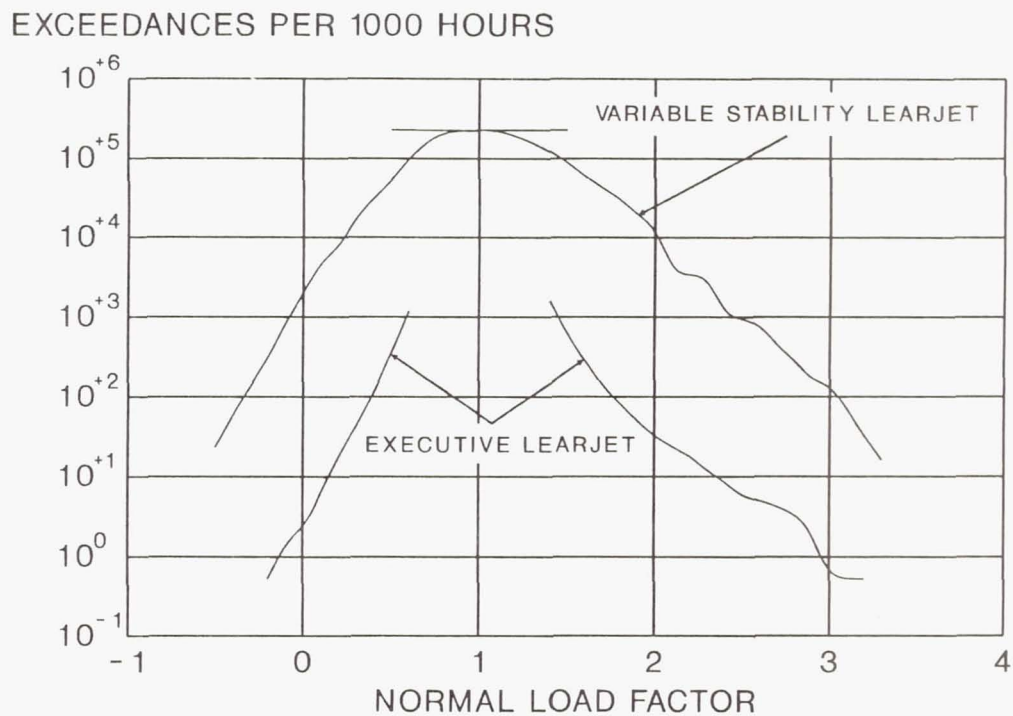


Figure 6. - Load Factor Exceedance for the V/S Learjet



# EXCEEDANCES PER 1000 HOURS

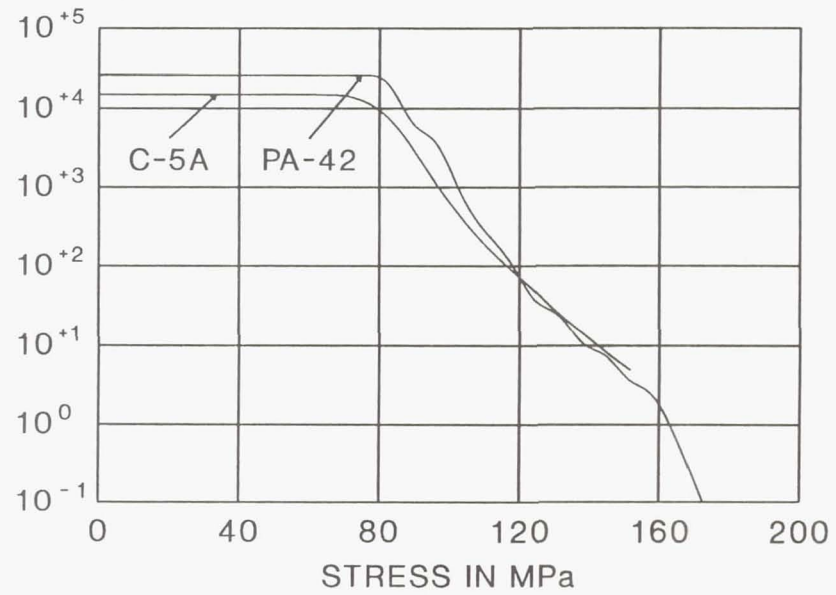


Figure 7. - Stress Exceedance Comparison for C-5A and PA-42

# INITIAL CRACK SIZE EQUAL TO 1.27 MILLIMETERS

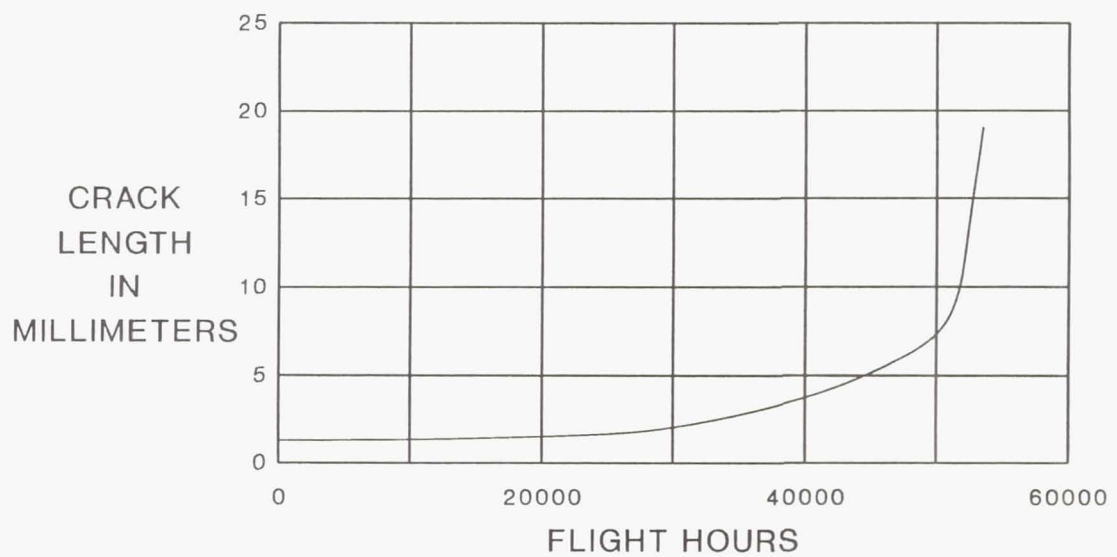


Figure 8. - Crack Growth Function for PA-42 - Aluminum

INITIAL CRACK SIZE EQUAL TO 1.27 MILLIMETERS

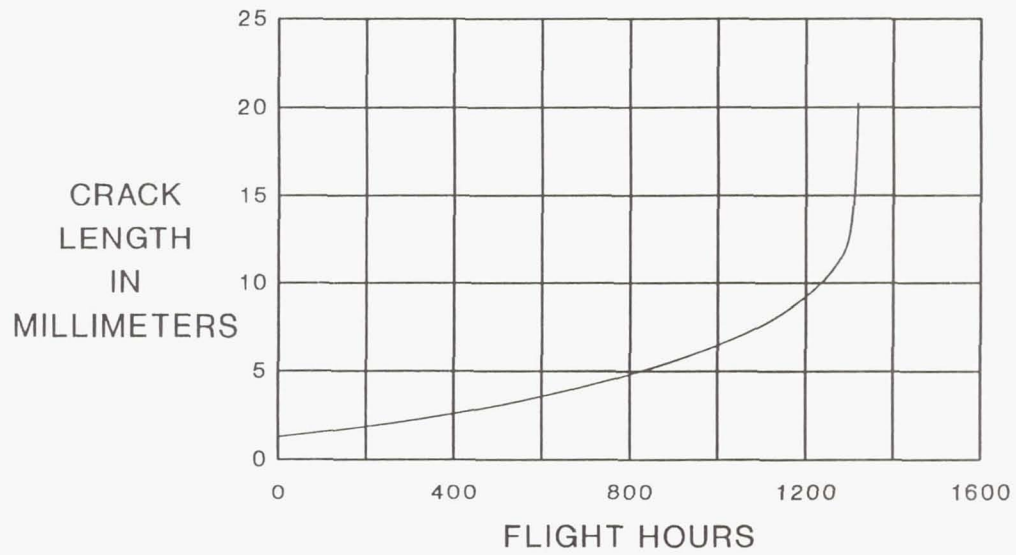


Figure 9. - Crack Growth Function for PA-42 - Steel

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Damaged Stiffened Shell Research at  
NASA Langley Research Center

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### Introduction

The structural behavior of stiffened fuselage shells with local cracks is influenced by the local deformations and stress gradients near the cracks, and this behavior must be understood to predict fuselage structural integrity and residual strength. As a crack grows, the stiffness and internal load distributions in a stiffened shell change, affecting the local deformations and stress gradients near the crack. Transport fuselage shells are designed to support internal pressure and mechanical loads which cause a geometrically nonlinear response. Research is being conducted at NASA Langley Research Center to develop and demonstrate a verified nonlinear stiffened shell analysis methodology for accurately predicting the global and local behavior of nonlinear stiffened fuselage shells with local cracks and with combined internal pressure and mechanical loads. The nonlinear structural analysis methodology being developed and planned at NASA Langley Research Center for damaged stiffened fuselage shells and the experiments being planned to verify this analysis methodology are summarized in the present paper.

### Analysis and Modeling

The nonlinear stiffened shell analysis methodology being developed for damaged stiffened fuselage shells includes developing analytical methods, developing a hierarchical structural modeling strategy, and conducting analytical studies of stiffened shells with cracks. The analytical methods development effort includes adding crack growth and residual strength analysis capabilities to the STAGS analysis code (ref. 1) for nonlinear structural analysis of general shells. The stiffened shell hierarchical structural modeling strategy is a global-local modeling strategy that provides sufficient modeling details to predict accurately the global nonlinear response of the shell and the local stress and displacement gradients that occur in stiffened shells with cracks. The analytical studies include

detailed nonlinear analyses of stiffened shells with cracks and with combined internal pressure and mechanical loads. The analytical studies also include parametric studies to determine the effects of varying structural and crack parameters and loading conditions on the behavior of stiffened shells with cracks.

The STAGS shell analysis code was originally developed to analyze the geometrically nonlinear response of shells with general geometries, boundary conditions and loading conditions. The code can be used to conduct strength, stability and collapse analyses of general shells with discontinuities such as a cutout in the shell wall. STAGS is a finite element analysis code that uses Newton's method for a nonlinear solution algorithm. Large rotations are included in the nonlinear solution by a co-rotational algorithm applied at the element level. The Riks arc-length projection method is used to integrate the nonlinear solution past limit points in the nonlinear solution. A crack propagation analysis capability is being added to STAGS for both self-similar and non-self-similar cracks. A load-relaxation capability has been developed to maintain equilibrium in a nonlinear state as a crack propagates in the model and causes the internal load distribution to change. An adaptive mesh refinement capability is being added to STAGS to provide the necessary modeling capability near a crack tip to represent accurately the stress and displacement gradients associated with a propagating straight or curved crack.

The modeling strategy being developed for the nonlinear analysis of stiffened shells with damage is a hierarchical modeling strategy that is intended to represent the important global and local features of the shell structure. Five hierarchical modeling levels are being evaluated to determine the modeling fidelity required to represent properly all important features of a stiffened shell without generating unnecessarily large models and unnecessarily expensive computational demands. These hierarchical modeling levels range from a relatively coarse global shell model for determining the internal load distribution and global response of the shell to a highly refined local detailed stress analysis model for determining crack growth characteristics from fracture mechanics principles. These hierarchical modeling levels are represented in figure 1. At each level, the results from the analysis of a more global model are used to define the boundaries of a more local detailed model which is more highly refined than the global model. The boundary conditions for each successive local model are kinematic constraints that are based on the results of the analysis of the more global model. The global shell model is used to determine



the global behavior of the shell with one or more cracks and with internal pressure  $p$  and mechanical loads that represent a bending moment  $M$ , a vertical shear load  $V$ , and a torsional load  $T$  applied to the shell. A nonlinear analysis of the global shell model provides the internal load distribution for the shell and identifies an interior local region that contains all the gradients associated with the crack. The boundary of this interior region is used to define the boundary of the stiffened panel model which is modeled in more detail than the coarser global shell model. The stiffened panel model is used to determine more accurately the interaction between the skin, frames and stiffeners as a crack propagates in the shell. The stress resultants  $N_x$ ,  $N_\theta$ , and  $N_{x\theta}$ , stiffener forces  $P_F$  and  $P_S$ , and boundary displacements and slopes are consistent with the global shell model results. The results from the stiffened panel model are used to define the local panel detailed model which is used to determine the interaction of a crack and a structural detail such as a lap splice. The results from the local panel detailed model are used to define a global-local panel model which is used to determine the detailed stress and deflection gradients near a crack. These gradients are used in the local detailed stress analysis model to determine crack growth behavior from a fracture mechanics analysis. The results of the crack growth analysis at the local detailed stress analysis model level are used to redefine the crack length and shape in the global shell model. The analyses at all hierarchical levels are repeated iteratively until the crack growth history and shell residual strength are determined. The attenuation lengths of all gradients are examined after each analysis to make sure that the model boundaries are far enough away from any gradient to avoid any boundary interaction effects. If a crack grows enough for its gradients to interact with a boundary, the appropriate local model is enlarged enough to prevent the interaction from occurring.

This hierarchical modeling strategy makes it possible to determine the detailed nonlinear response of a large stiffened shell model with cracks and with combined internal pressure and mechanical loads. The global response and local response phenomena such as local instabilities, modal interaction and stiffener distortion and rolling can be determined. The effects of crack propagation on the nonlinear shell response can also be determined. The changes in the local stiffnesses and internal load distribution in the shell can be determined as the crack grows. The coupling between inplane and out-of-plane deflection gradients and



internal forces and moments are accurately represented in the nonlinear analysis and changes in crack growth direction can be determined.

The hierarchical modeling strategy also makes it possible to determine the effects of changing global and local shell model parameters on nonlinear response, crack growth characteristics and residual strength. The modeling strategy allows the effects of varying shell parameters such as shell radius, skin thickness, frame spacing and stringer spacing to be studied. Structural detail parameters such as frame and stringer geometries, joints, splices, repairs, failsafe straps and clips can also be modeled and studied. Different combinations of internal pressure and mechanical loads can also be studied. Skin crack location, length, orientation and propagation direction can be studied as well as broken frames and stringers. The interaction between skin, frame and stringer as a crack propagates can also be determined.

### Analytical Results

A limited number of analytical results have been generated to determine the modeling and analysis requirements for a stiffened shell segment representative of a generic narrow-body transport fuselage. The shell model is 240 inches long and has a 74 inch radius, a 0.036 inch skin thickness, a 20 inch frame spacing and a 9.3 inch stringer spacing. The shell is made of aluminum and has flat end caps and floor beams.

Linear and nonlinear analyses were conducted with a global shell model without a crack to determine the differences in shell response up to 8 psi of internal pressure. The stress distribution and local bending in the skin at the skin-frame intersections are different for the two analyses indicating the influence of nonlinear behavior on the results. The hoop stress resultant distribution for these analyses are shown in figure 2. Nonlinear analyses were also conducted for global shell models having a 20-inch-long skin crack, with and without a broken frame at the center of the crack, to determine the shell response up to 8 psi of internal pressure. The crack is located near a stringer at the fuselage crown and is bisected by the frame. The hoop stress resultant distribution for these analyses is shown in figure 3. The effects of the cracks on the hoop stress resultants can be determined by comparing the nonlinear results in figures 2 and 3. The differences in the hoop stress resultant distributions indicate the extent of the effects of the skin

crack and broken frame. The hoop stress resultant gradients associated with the cracks extend over three frames, four skin bays between frames, and seven stringers. The radial displacement gradients in the skin near the 20-inch-long skin crack and broken frame are shown along the stringer and along the arclength of the broken frame in figures 4 and 5, respectively. The skin radial displacement is normalized by the skin thickness in the figures. The results for the damaged shell are represented by the solid line in each figure and the results for the undamaged shell are represented by the dashed line. The maximum values of the radial displacements shown in figure 4 occur at longitudinal stations on the model corresponding to locations indicated in the figure as the axial distances along the stiffener at  $x = 55$  in. and  $x = 65$  in. The radial displacement at the location corresponding to  $x = 60$  in. is a lower value than the nearby maximum values because the broken frame at  $x = 60$  in. is stiff enough to prevent a larger radial displacement at that location. The maximum value of the radial displacement shown in figure 5 occurs at a circumferential station on the model corresponding to a location just to the right of the location indicated in the figure as arclength along the frame at  $s = 70$  in. The displacement at the location in the figure just to the left of  $s = 70$  in. is smaller than the maximum displacement because the stringer is attached to the skin to the left of the crack and the stringer is stiff enough to prevent larger displacements for this part of the skin.

Comparisons of radial displacement and hoop stress resultant results from linear and nonlinear analyses of the fuselage shell with a 20-inch-long skin crack and a broken frame are shown in figures 6 and 7, respectively, for internal pressures up to 8 psi. The linear results are represented by the solid lines and the nonlinear results are represented by the dashed lines. The results for the shell without a crack are represented by the circles and the results for the shell with the crack are represented by the squares. The radial displacements in figure 6 are measured at the middle of the crack and are normalized by the skin thickness. These results indicate that the radial displacements in the shell without the crack are larger at this location for the nonlinear analysis than for the linear analysis. Also, the radial displacements in the shell with the crack are smaller at this location for the nonlinear analysis than for the linear analysis. The hoop stress resultants in figure 7 are for the finite element just ahead of the crack tip. These results indicate that the hoop stress resultant in the shell without the crack is larger at this location for the linear analysis than for the nonlinear analysis. Also, the hoop stress resultant in the shell with the crack



is larger at this location for the nonlinear analysis than for the linear analysis. These differences in results indicate that the conventional view that a linear analysis is a conservative analysis for predicting the response of a nonlinear stiffened shell with internal pressure is not always appropriate. Nonlinear analyses are needed to provide accurate predictions of the behavior of such shells.

Radial displacement results from a nonlinear analysis of a stiffened panel model with internal pressure of 8 psi are shown in figure 8. The skin, frames and stiffeners are modeled with plate finite elements and the deformations of the skin and the frames are shown in the figure. The results of this stiffened panel analysis helped guide the development of a local panel detailed model of the skin bay between two stringers and two frames with a symmetric mid-bay skin crack. A refined finite element mesh was used for the model of the skin bay region where the crack is located. The undeformed model and the hoop and axial stress resultants from a nonlinear analysis of the skin bay without a crack are shown in figure 9 for 8 psi of internal pressure. The corresponding results for the skin bay with a 1-inch-long and a 2-inch-long mid-bay skin crack are shown in figure 10. The distribution of the hoop and axial stress resultants near the crack tip indicates large increases in stresses near the crack discontinuity. The mid-bay radial displacements for the skin without a crack and with 1-inch-long and 2-inch-long cracks are shown in figure 11. The circles represent the results for the skin without a crack and the triangles and squares represent the skin with the 1-inch-long and 2-inch-long cracks, respectively. These results indicate that increasing the crack length from 1 inch to 2 inches causes a significant growth in the radial displacements near the skin crack.

### Planned Experimental Facilities

Experiments are planned at NASA Langley Research Center to verify the structural analysis methodology being developed and to identify critical behavioral characteristics that must be understood. The experiments will be conducted on curved stiffened panels and stiffened shells representative of current transport aircraft fuselage structures. The planned experimental program will include specimens that will be subjected to combinations of internal pressure and mechanical loads and different damage conditions.

Two types of test fixtures are being developed for the NASA Langley Research Center structural test program and these test fixtures are shown schematically in figure 12. A pressure box



fixture will be used to test curved stiffened panels that are subjected to internal pressure and biaxial hoop and axial loads. Hoop stress resultants  $N_\theta$  and frame forces  $P_F$  will be generated as reactions to the applied internal pressure  $p$ . Axial stress resultants  $N_x$  and stringer forces  $P_S$  will be generated by hydraulic jacks. A closed-cell combined-load test fixture will be used to test curved stiffened panels and stiffened shells that require internal pressure, bending  $M$ , and torsion  $T$  loads. This fixture will allow hoop and axial stress resultants and shear stress resultants  $N_{x\theta}$  to be applied to the specimen as well as internal pressure. Hydraulic jacks will be used to generate the mechanical loads for the combined-load test fixture.

### Test Results

A NASA-owned 737 transport fuselage was instrumented and pressurized to 6 psi internal pressure to determine the strains and radial displacements near a longitudinal lap joint in the fuselage skin. The results of this test are presented in reference 2 and the radial displacements of the skin along the circumference midway between a frame and a tear strap are shown in figure 13. Eleven differential voltage displacement transducers were used to make these measurements. The results in figure 13 show the radial displacements in two skin bays located between three adjacent stringers. The stringers are 9.5 inches apart and the lap joint is at the middle stringer. The results indicate that relatively large displacements occur in the skin between these stringers and noticeable bending gradients are present in the skin. The maximum relative displacement is on the order of the skin thickness which indicates that geometric nonlinearities should be included in the corresponding shell analysis.

### Concluding Remarks

A verified structural analysis methodology for stiffened fuselage shells with damage is being developed at NASA Langley Research Center. This structural analysis methodology is intended to represent the nonlinear behavior of damaged stiffened shells with combined internal pressure and mechanical loads. Crack growth and residual strength analysis capabilities are being added to a nonlinear structural analysis code for general shell structures. A hierarchical global-local modeling strategy is being developed to represent

accurately the nonlinear global response of a stiffened shell subjected to combined loads and the local stress and displacement gradients in shells with propagating cracks. Analytical studies of stiffened shells with cracks and with combined internal pressure and mechanical loads are being conducted to determine the effects of varying structural and crack parameters on the behavior of stiffened fuselage shells with damage. The structural analysis methodology will be verified by experiments on curved stiffened panels and stiffened shells subjected to internal pressure and mechanical loads. Tests and facilities are planned for investigating the behavior of stiffened panels and shells with damage and combinations of internal pressure, bending, and torsional loads that generate the inplane normal stress resultants and shear stress resultants representative of transport fuselage loading conditions.

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1. Almroth, B. O.; and Brogan, F. A.: The STAGS Computer Code. NASA CR-2950, 1980.
2. Phillips, Edward P.; and Britt, Vicki O.: Measurements of Fuselage Skin Strains and Displacements Near a Longitudinal Lap Joint in a Pressurized Aircraft. NASA TM 104163, October 1991.

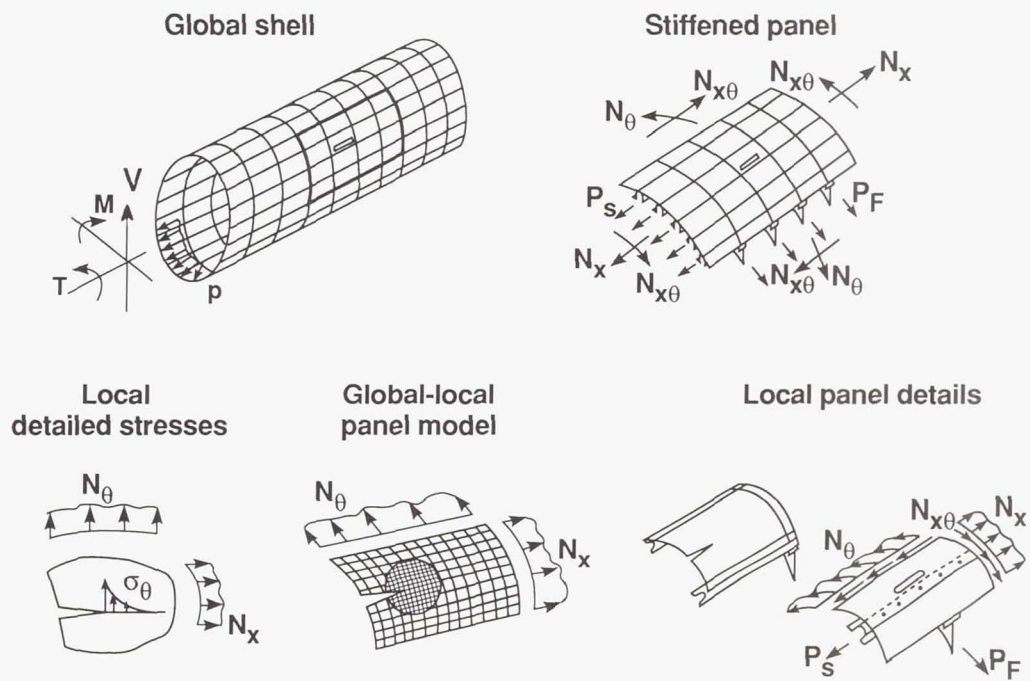


Figure 1. Hierarchical nonlinear stiffened shell models.

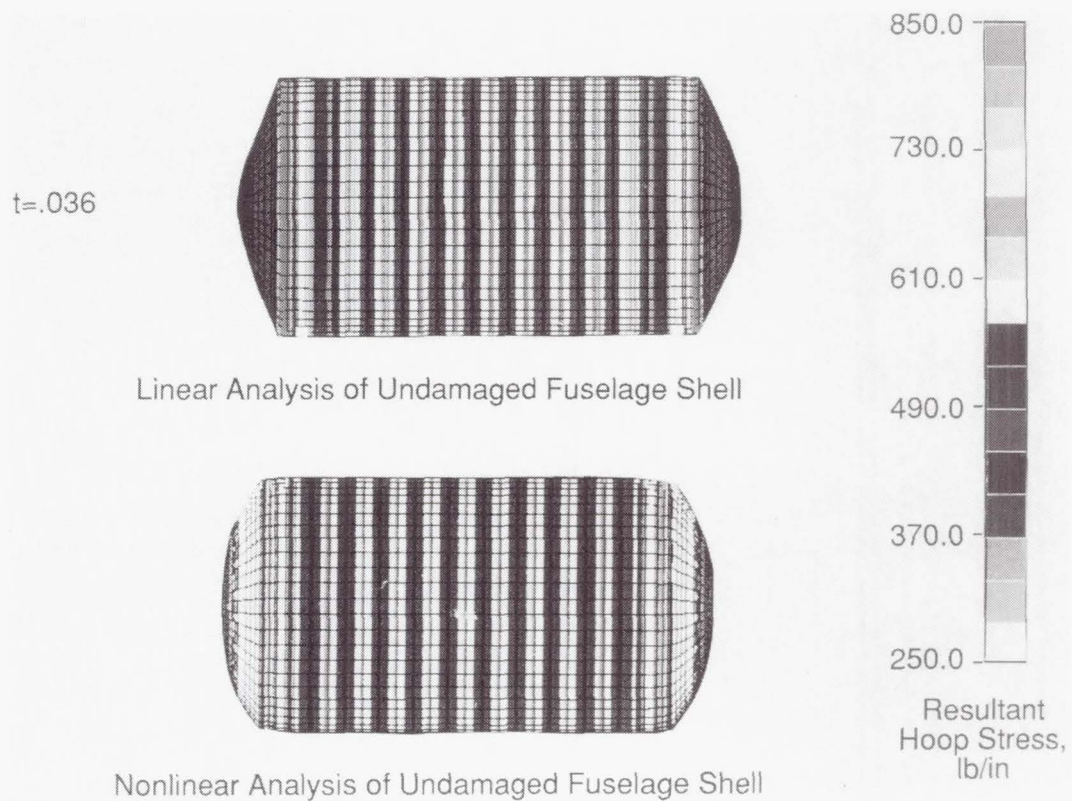


Figure 2. Hoop stress resultant distribution for a stiffened shell with 8 psi of internal pressure.



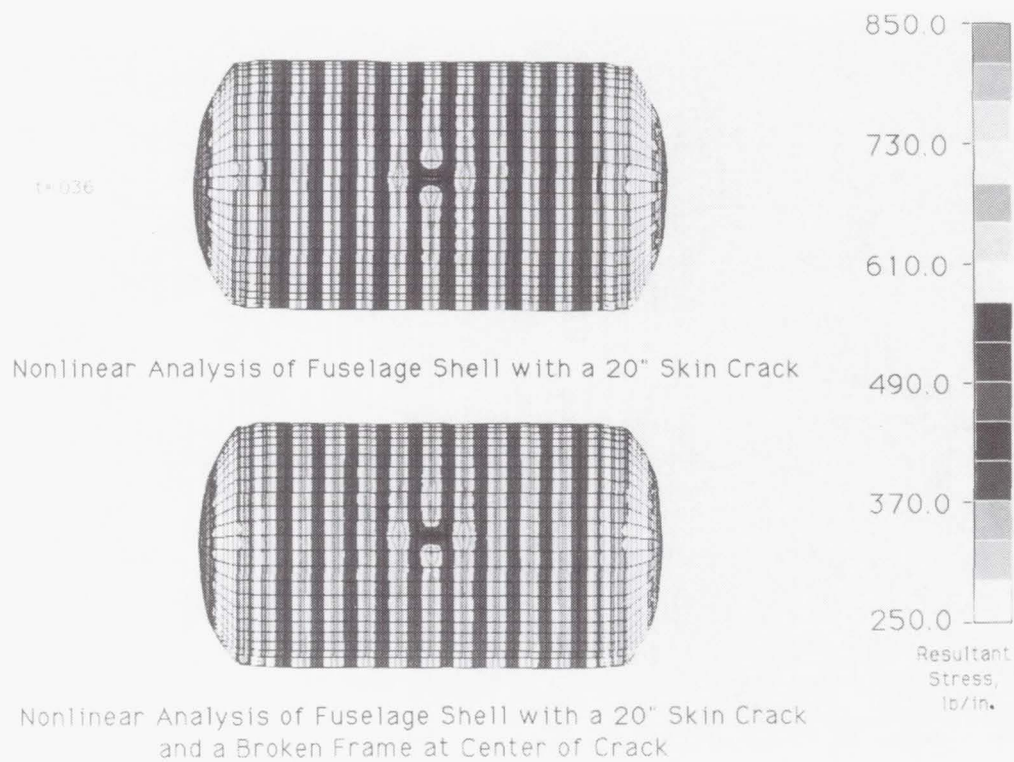


Figure 3. Hoop stress resultant distribution for a stiffened shell with 8 psi of internal pressure and a 20-inch-long longitudinal skin crack, with and without a broken frame.

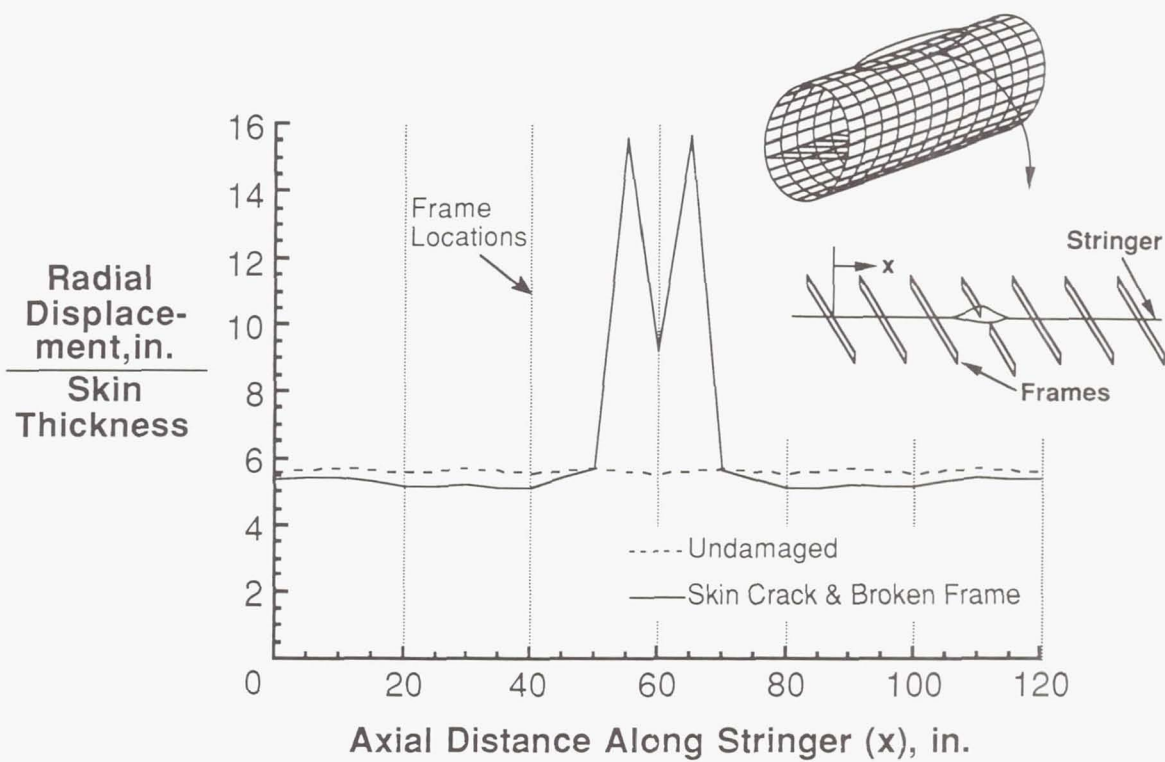


Figure 4. Radial displacements of the skin along a stringer near a skin crack and a broken frame in a stiffened fuselage shell structure. Results are from a nonlinear analysis with 8 psi of internal pressure.

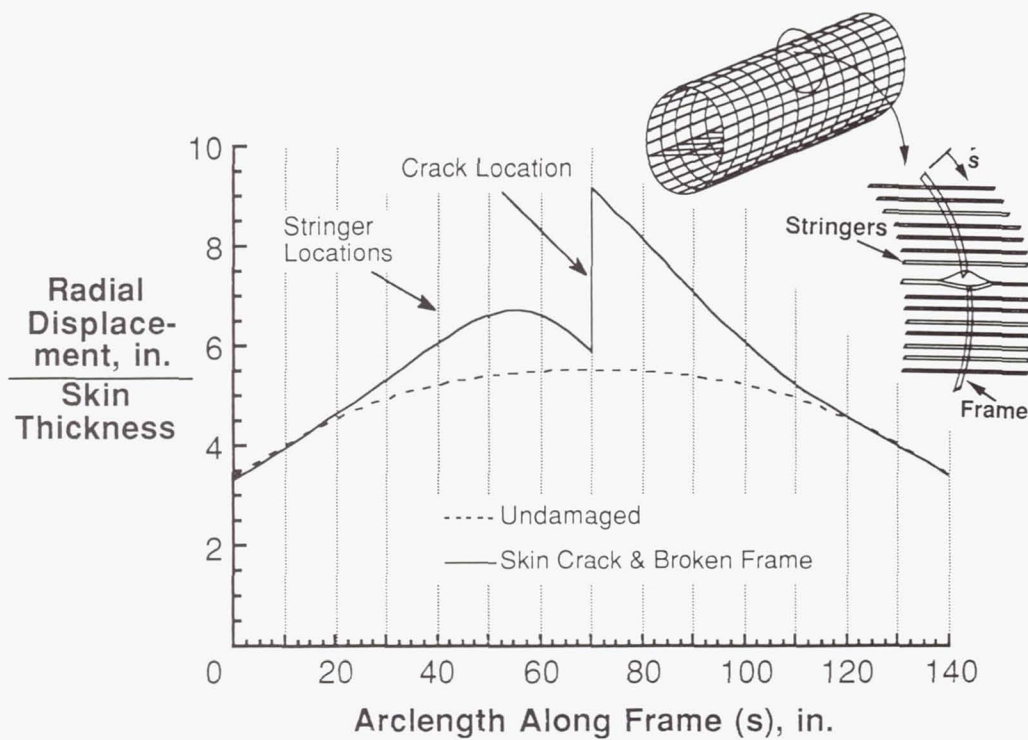


Figure 5. Radial displacements of the skin along a broken frame in a stiffened fuselage shell structure. Results are from a nonlinear analysis with 8 psi of internal pressure.



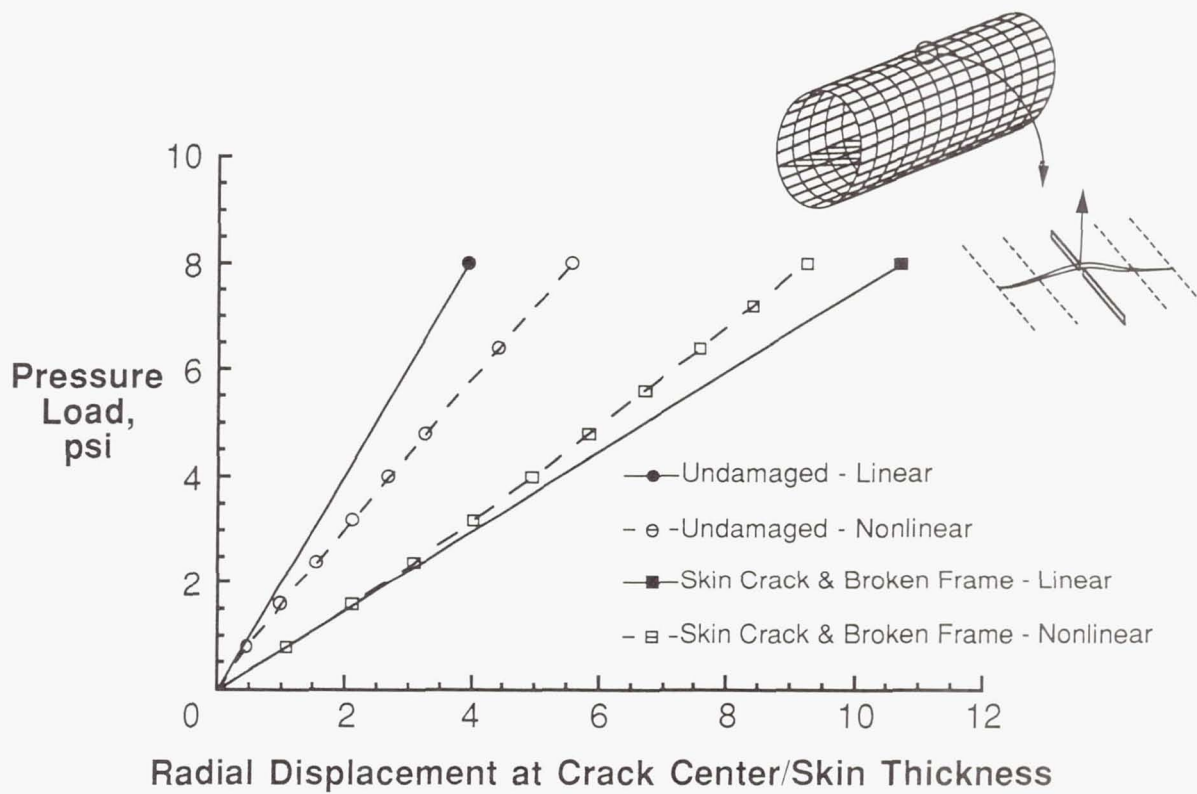


Figure 6. Effect of increasing internal pressure on the radial displacements of a fuselage skin at a frame.

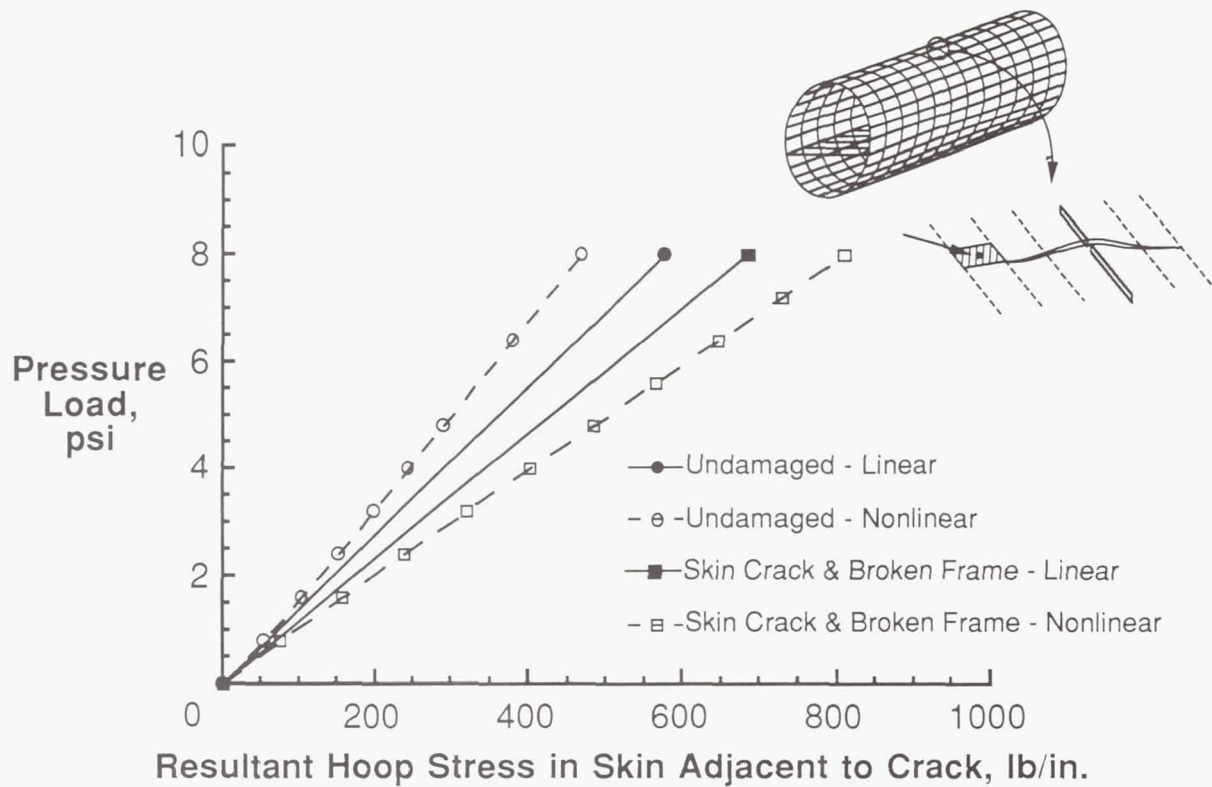


Figure 7. Effect of increasing internal pressure on the hoop stress resultant in a fuselage skin near a crack tip.

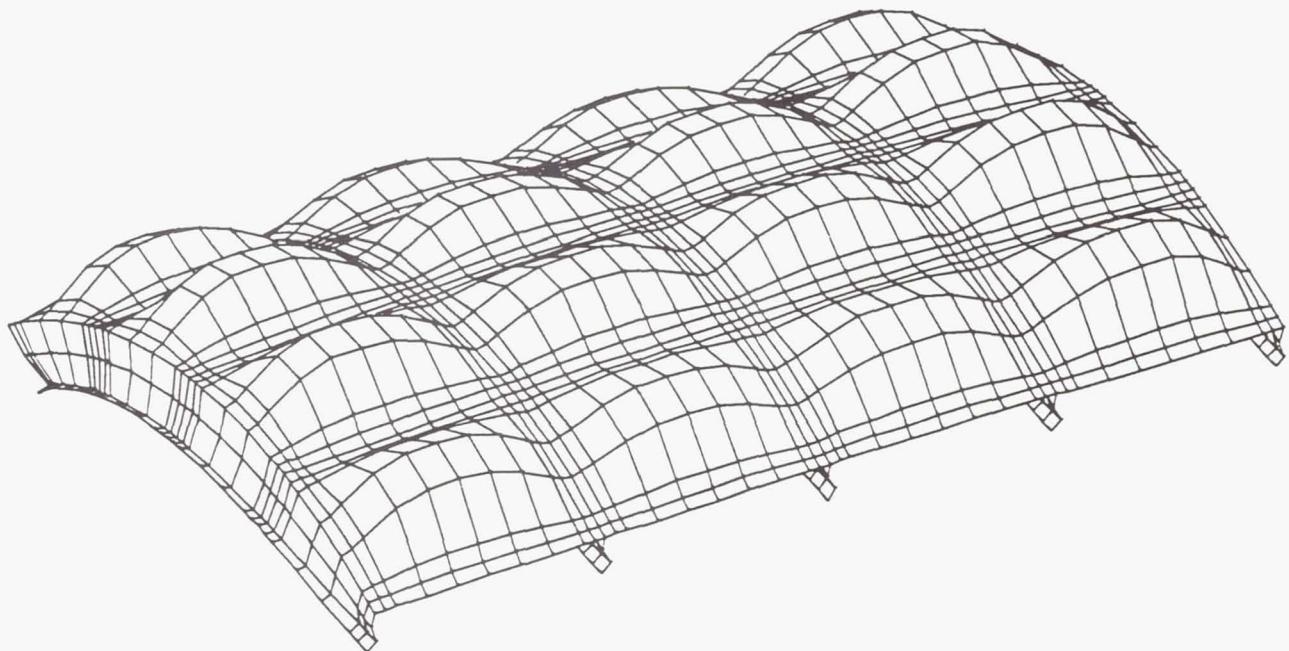


Figure 8. Radial displacements in a stiffened panel with 8 psi of internal pressure.

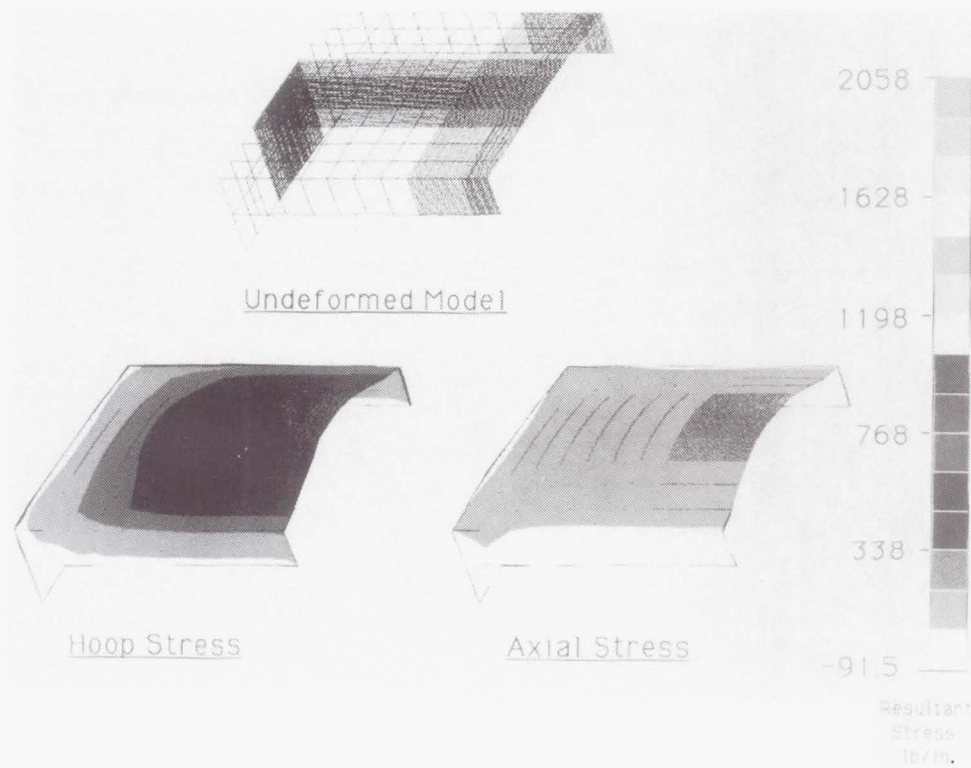


Figure 9. Undeformed model and hoop and axial stress resultants in an undamaged skin bay of a stiffened fuselage shell.



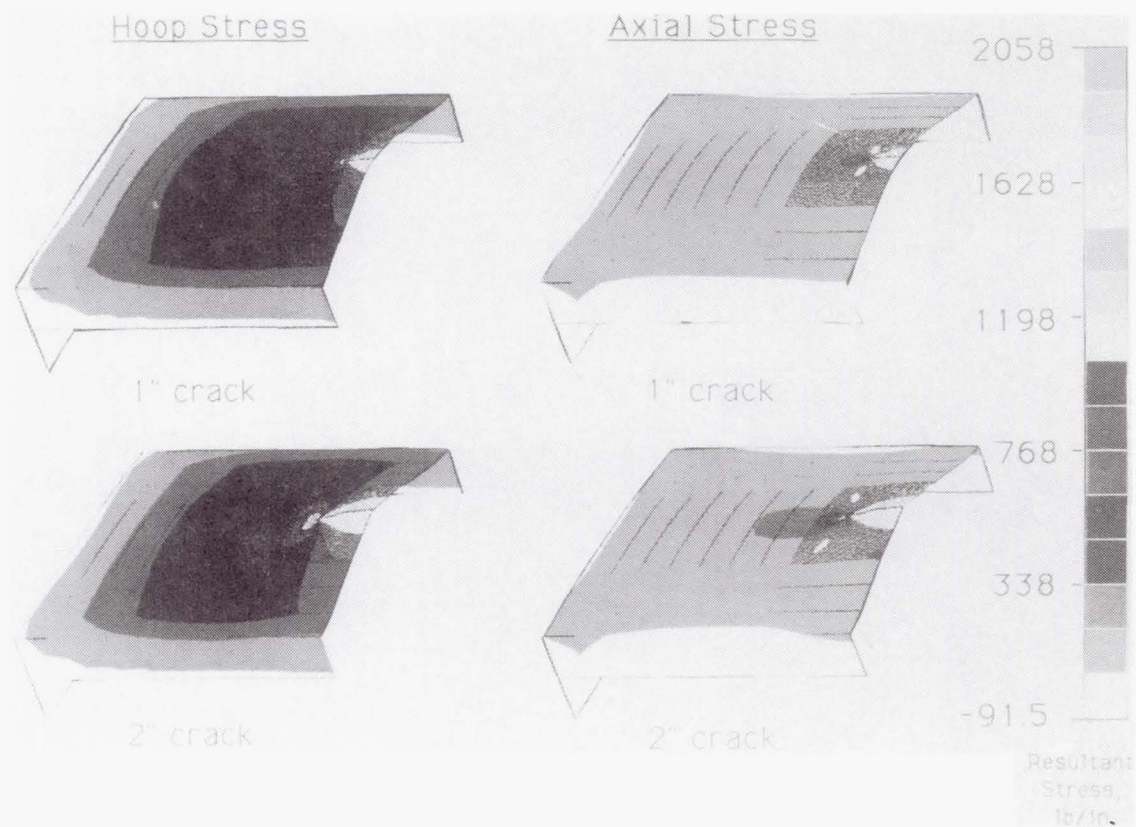


Figure 10. Effect of increasing crack length on the hoop and axial stress resultants in a skin bay of a fuselage shell with mid-bay skin cracks.

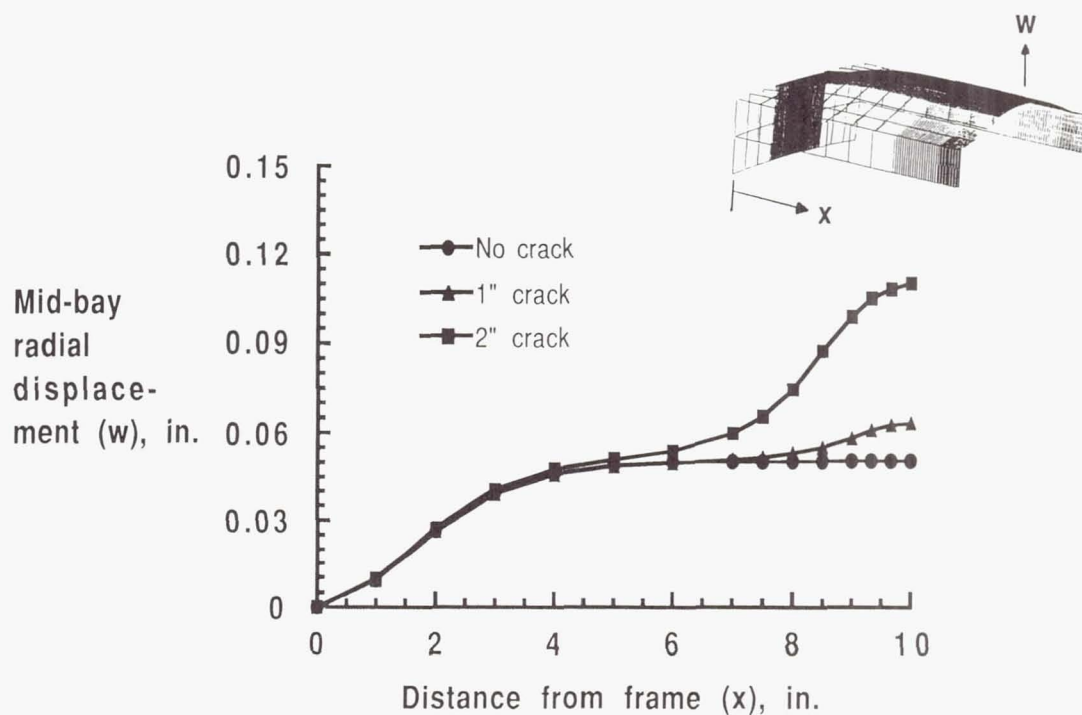
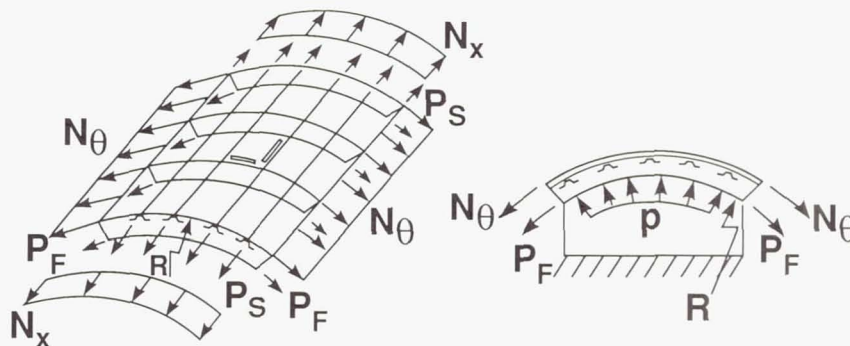
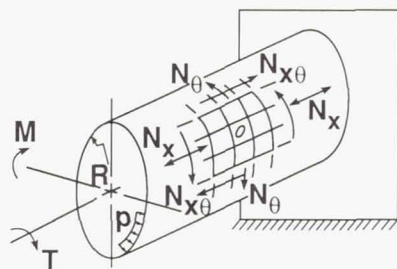


Figure 11. Longitudinal radial displacement distribution along the mid-bay of the skin of a stiffened fuselage shell with and without skin cracks.

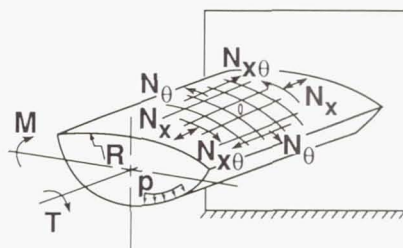
### Curved-panel pressure box for biaxial loads



### Combined pressure-bending-shear load fixtures



Cylindrical shells/panels



D-box panels

Figure 12. Curved stiffened panel and stiffened shell test fixtures.

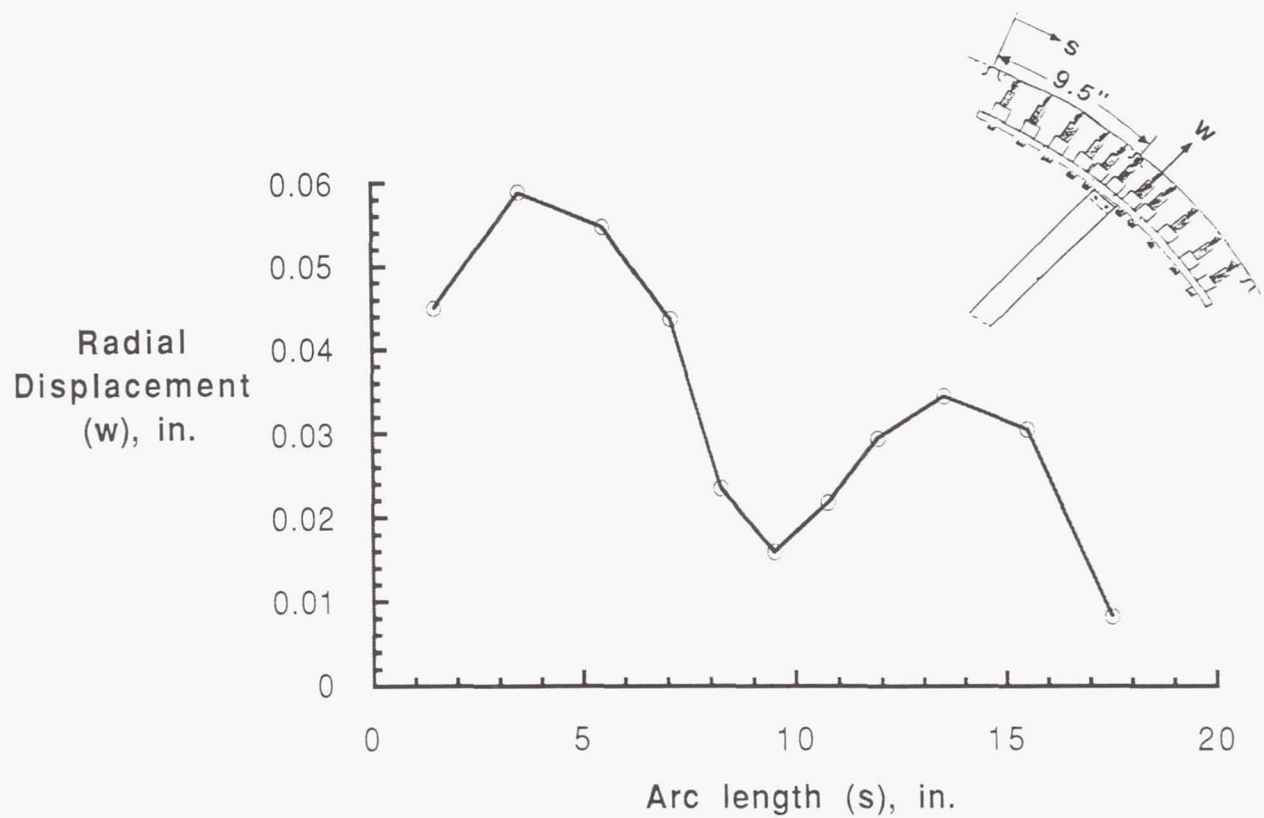


Figure 13. Measured radial displacement in the skin of a stiffened fuselage shell with a longitudinal lap splice and with 6 psi of internal pressure.



# FEDERAL AVIATION ADMINISTRATION AGING AIRCRAFT

## NONDESTRUCTIVE INSPECTION RESEARCH PLAN

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### SUMMARY

This paper highlights the accomplishments and plans of the Federal Aviation Administration (FAA) for the development of improved nondestructive evaluation (NDE) equipment, procedures, and training. The role of NDE in aircraft safety and the need for improvement are discussed. The FAA program participants, and coordination of activities within the program and with relevant organizations outside the program are also described.

### BACKGROUND

Airplanes built prior to 1978 were designed to the fail safe principles, but with no specific damage growth or damage tolerance requirements. Since 1978 all new transport aircraft are required to be damage tolerant. Newly designed aircraft must be designed as damage tolerant and aircraft designed before the regulation was promulgated are required to have manufacturer defined programs to make them damage tolerant. Succinctly, the damage tolerance philosophy states: cracks are expected to occur; the structure must tolerate "small" cracks; and cracks must be detected before they become "large". In order to implement this regulation, manufacturers were required to define inspection programs for each of the "probable location and modes of damage". Some key points to note: cracks are expected to occur; cracks are expected to grow unacceptably large if not detected; and damage tolerance requires reliable crack detection.

Although the inspection system has generally worked well, aging aircraft are straining it. Presently, about 30% of the transport category fleet is greater than 20 years old. It is estimated that by the year 2000, greater than 60% of the transport category fleet will be more than 20 years old. Several issues have arisen which cause concern.

- Multiple Site Damage (MSD) - numerous, small, interacting cracks
- Unpredicted Crack Sites - cracks in unexpected, and therefore uninspected places
- Inaccessible Areas - areas which are difficult to inspect due to lack of access

The concern over MSD has led to mandated terminating actions to replace rivets and increased inspection intervals. These actions are extremely costly to the industry. A more reliable means of detecting inaccessible second layer cracks and corrosion is needed, especially as regards to repair

patch configurations. In addition, the commuter fleet has come under increased awareness of crack detection and corrosion reliability needs. Visual inspection continues to be the predominant, most practical choice for aircraft inspections in the field. Therefore, high priority and great emphasis must be placed on enhanced and augmented visual inspection techniques. Especially in the commuter fleet application, NDI developments must be noncomplex and inexpensive to be cost effective and implementable. Due to these pressures on the detection system, improved inspections are required. Input from airframe manufacturers confirms that in general, equipment and procedures currently in use should be improved such that smaller defects are detected, any indications are well documented, and disturbances are eliminated so that the false alarm rate is low.

Each of six factors that affect airworthiness assurance: corrosion; fatigue and fracture; maintenance and repair; flight loads; human factors; and nondestructive inspection (NDI) are subprograms of the overall Federal Aviation Administration (FAA) National Aging Aircraft Research Program (NAARP). Inspection is the vehicle that identifies damaged aircraft, and sends them back for repair, as shown in Figure 1. Other elements of the system can be improved to slow the rate at which deterioration occurs. However, inspection is the only element which can immediately improve the safety of the system by finding non-airworthy aircraft.

The term NDI denotes the means for establishing the quality or integrity of materials and structures without impairing or affecting their end use. To this end, NDI has provided the basis for essential quality assurance and maintenance inspection criteria in the aircraft industry. NDI methods are used to supplement basic visual inspection to: avoid costly tear down in gaining access to hidden structure; provide early detection of defects before they reach critical size; and obtain additional information. In many situations, there is no practical alternative to NDI procedures. NDI procedures are mandatory for some structural inspections to support supplemental structural inspection programs and to support service bulletin and airworthiness directive call outs. The six most commonly used NDI methods are: visual; eddy current; radiographic; ultrasonic; penetrant; and magnetic particle. Their applications and possible new technology improvements are listed in Figure 2.

## PROGRAM MANAGEMENT

The NDI portion of the NAARP predicates that improvements must be made to existing inspection techniques and devices, and research and development efforts should be directed towards new and emerging technologies, so as to meet the challenge of more reliable detection capabilities. The objectives for the NDI program are derived directly from the concerns about the current system, as exemplified by the factors which led up to the Aloha accident. These objectives are closely aligned with the priorities defined by the Air Transport Association in 1988. The FAA program seeks to improve crack detection reliability, develop capabilities for scanning broad areas of the structure to detect damage, develop means to reliably detect and differentiate various forms of corrosion, develop techniques for measuring bond quality and bond strength, and identify inspection protocols that would ensure consistently high inspector performance.

The program is making use of several resources to achieve the objectives. These resources have



been integrated into a single management structure, as depicted in the attached Figure 3, to assure: a mix of near, mid, and long term research; integration of knowledge from organizations outside the FAA's purview; allocation of funds to most efficiently reach program objectives; and mechanism for encouraging and integrating innovative technologies.

To this end, an Aging Aircraft NDI Working Group (AANWG) consisting of NDI experts from government, industry and academia has been established. AANWG was specifically set up to advise the program office about the technical merit of various ideas which may be proposed. The FAA National Resource Specialist (NRS) advises the program office on the direction of the program and chairs the AANWG group.

In addition, the National Aeronautics and Space Administration (NASA) which has run a large in-house NDI program for some time, primarily oriented towards the space shuttle, has agreed to maintain a close liaison between the two agencies through a formal memorandum of understanding which assures that projects are complimentary, with little overlap. A two way flow of technical information has been established benefiting both organizations. For example, workshops are being planned for joint sponsorship by the FAA and NASA this winter on topics like infrared imaging and ultrasonics. NASA will continue to focus on basic research in NDI, especially in the area of broad or large area aircraft inspection techniques, while the FAA program addresses continued airworthiness issues.

Because of the diversity of the overall nondestructive inspection requirements, the FAA will utilize the abilities of various technology centers in support of the different program tasks. A continuing effort is underway to identify and integrate universities and research institutes into our program structures to help execute the NAARP. Iowa State University, Sandia National Laboratory, Lawrence Livermore National Laboratory, Harwell Laboratory, and Carnegie-Mellon University have all so far been selected for inclusion to the Program. Starting in FY-93, the Program budget requests provide the funding necessary to fully integrate these and possibly several more participants into the integrated Program structure.

## RECENT ACCOMPLISHMENTS

Iowa State University has established a Center for Aviation Systems Reliability (CASR) with congressionally directed funds to pursue applied basic research in support of the FAA Aging Aircraft Research Program through a consortium which includes Northwestern and Wayne State Universities. The purpose of this group is to focus specialized university talents on application of NDI fundamentals to concepts which could be developed, implemented and integrated into the aircraft inspection system. Initial projects and tasks have been defined for CASR in the areas of characterization of adhesive bonds, large scale inspection technique development, neural nets and expert systems, in-situ sensor development, characterization of materials, and image analysis for future system automation.

In FY-91, \$3M was obligated to the CASR to begin work on the above tasks through an interagency agreement with the Department of Energy (DOE). An additional \$3M has been transferred to DOE by the FAA and negotiations are underway with the CASR for defining tasks



from the CASR for a Congressionally directed \$1.5M laboratory expansion design to assure that specialized university talents in basic research and education and training as related to aging aircraft problems are applied in the long term NDI program.

Sandia National Laboratory, in support of the FAA Aging Aircraft Research Program, has established an aging aircraft NDI Development and Demonstration Center (AANC) through a consortium led by Sandia National Laboratories which includes Science Applications International Corporation (SAIC) and New Mexico State University. Their role is to develop concepts which come from CASR/NASA research, and to demonstrate and evaluate new technologies. "Hands-on" experience for field personnel will be afforded, important for transferring technology to the field.

Technology transfer will be accomplished by choosing proven or emerging NDI technologies capable of solving current aircraft inspection problems, adapting them for field inspection, verifying their capabilities, and transferring them to the aircraft industry.

In August, \$3.4M in congressionally directed funding initiated the efforts at Sandia National Laboratories for FY-91 and FY-92. The major task being undertaken by Sandia with this initial funding is to determine how well current equipment and procedures being used in the field can find cracks of a given size under a given set of circumstances. Work to determine probability of detection (POD) has been done at major aircraft manufacturers, including recently, Boeing Commercial Airplane Company. However, the experiments have been performed in a quasi-laboratory setting, and have not taken into account the problems of environmental and human factors influencing POD (such as physical access difficulty and boredom). Also, no attempt has been made to optimize instrumentation and procedures. Three pilot projects have been identified by the FAA: POD of cracks in lap joints in transport aircraft; POD of cracks in commuter aircraft inspection scenarios; and POD of visual inspection of cracks.

These projects are looking at eddy current, and/or visual inspection as a process to be analyzed under field conditions. The analysis will determine the probability that inspectors could find appropriately sized cracks and make suggestions for improving the inspections. The five year agreement created with Sandia establishes a long term NDI prototype instrumentation and technique validation facility supported by: maintaining a library of characterized samples for use in technique validation; developing and operating test beds for NDI technique development and validation; and developing and maintaining a standard data base for validation test data.

Harwell Laboratory is assisting Sandia in establishing a validation tool which can quantitatively and independently assess the effectiveness and reliability of newly developed and existing inspection methodologies. A specific objective of the work by Harwell will be to quantify the effects of human factors on the probability of detection of cracks. The work will measure human responses when required to perform repetitive inspections, and under varying work conditions.

FAA Technical Center in-house efforts have been directed at identifying areas in commercial aviation where emerging NDI techniques can be a cost-effective replacement for existing inspection procedures that are time consuming, labor intensive, and not totally reliable. Under a Cooperative Research and Development Agreement with Henson Aviation, a shearographic demonstration inspection of portions of specific interest of the fuselage of a Boeing 737-200 was



performed at the carrier's repair station at Winston-Salem, North Carolina. The demonstration indicated potential advantages of shearography over currently used ultrasonic inspection techniques for detection of disbonds in the fuselage, and reduced down-time of the aircraft with concomitant reduced inspection costs. USAir Express engineers present during test and evaluation have concurred with our draft findings based on carrier provided data that shows the potential of saving hundreds of labor hours and up to three days of aircraft down-time per inspection compared with respective times listed in pertinent Service Bulletin. Additional demonstrations on commuter type fuselage have been scheduled on a DeHavilland DHC-7 with USAir Express in December.

A magneto-optic imaging device based on unconventional eddy current excitation and a new method of extracting the desired flaw information, using a magneto-optic sensor to form images of flaws directly and in real time over relatively large areas, was evaluated for detection of cracks around rivet holes. Concurrently, Boeing, Douglas, and American Airlines also evaluated the device and American has received limited approval from the FAA for specific rivet inspection procedures. The Technical Center will work with the manufacturer to further evaluate the technique on additional fuselage applications such as second layer crack and corrosion detection.

A subcontracted JT8D series engine flight safety review was completed, which identified reliability problems through actuarial analysis of causes of in-flight shutdowns and engine removals. Development is ongoing for enhanced NDI procedures for the identified troublesome components with the cooperation of Pratt and Whitney engineers. Further development of critical components for CF6 and JT3D engines requiring enhanced NDI procedures is underway.

#### NEAR TERM DEVELOPMENTS

In order to assimilate additional industry expertise in NDI procedures and techniques into the Program, we are currently working on contracting mechanisms which can expedite the process for input from the private sector. In addition, several agencies within the United States and foreign governments possess strong NDI talent and are pursuing research of their own. It is expected that valuable cross-fertilization will result from cooperative agreements being established with these agencies.

For example, Lawrence Livermore National Laboratory (LLNL) has developed dual band infrared imaging methods and unique image correction algorithms for detection and characterization of subsurface geologic features. They have proposed to apply these methods for broad area scanning of airframes to provide early warning of subsurface flaws. These methods developed at LLNL provide high sensitivity through improved signal to noise and separation of surface emissivity effects from true thermal patterns. They will be working in cooperation with General Electric Aircraft Engine Company NDI specialists on a companion study to determine the feasibility of detecting low cycle fatigue cracks in aluminum airframe skins with infrared imaging.

The most widely used technique for field inspection of aircraft is visual inspection or enhanced visual inspection using dye penetrant or magnetic particles. Since this is the most widely used inspection technique, any improvement in this area would have a major impact on inspection quality. A number of areas of enhanced visual or optical inspection are being considered for



quality. A number of areas of enhanced visual or optical inspection are being considered for accelerated development and evaluation:

1. Detection of surface waviness conditions such as "pillowing" caused by corrosion may be enhanced by structured light, moire, or D-Sight (a simple optical arrangement involving a source of light and a retro-reflective screen).
2. A process using spectroradiometrics, employing color filters with optical devices such as glasses or borescopes, to detect colors associated with corrosion products.
3. Application of commercially available borescope and fiberscope technology to aircraft fuselage inspection behind skins through a removed rivet for corrosion detection.

In FY-91, a feasibility study was initiated at Carnegie-Mellon University in Pittsburgh, PA to design and develop robotic tools to assist the aircraft inspector and to automate the collection, archiving and post-processing of inspection data. Upon successful establishment of the feasibility of large-scale robotic inspection systems for use by major inspection and repair facilities this fall, Carnegie-Mellon will be funded to initiate their long range goal of robotic system or systems development to perform routine aircraft inspection tasks that are currently done manually. These robotic tools will be designed in coordination with guidance and recommendations provided by USAir maintenance and inspection experts from their Pittsburgh facility, so that the end results will be readily acceptable to aircraft maintenance hanger personnel.

Current FAA training programs including a two week training course, "Nondestructive Testing," which is used for training of both engineers and inspectors are being reviewed for possible updating to include technological advances, such as new NDI equipment and inspection procedures now being used by the aviation industry. A survey has been prepared in cooperation with the Flight Standards Service (AFS) and is being distributed to appropriate field personnel. Field data will also be gathered in visits to AFS selected sites in subject areas involving evaluation of non-destructive testing methods, choice of appropriate methods, application of individual methods, limitations and surveillance of methods. A final report will be prepared this spring that summarizes compilation of the survey data, field site analysis and evaluation of the training program. This will include recommendations for any necessary revisions to existing Academy training courses as well as suggestions for new training efforts.

In addition, development of a training video that provides instruction in nondestructive inspection of civil transport and commuter airlines for corrosion detection is underway. The videotape presentation will incorporate an in-service field demonstration of NDI equipment and techniques currently used by commercial operators and independent repair/maintenance facilities in inspection of civil transport and commuter airplanes. Techniques to be included are visual inspection, radiography, ultrasonic and eddy current techniques as they apply to corrosion evaluation and detection. A draft video presentation package will be available in January for AFS review.

To further develop, in all NDI program areas, the best evolution of technology into practical commercial aviation applications, an International NDI Workshop, with that theme, will be sponsored by the FAA, NASA, Navy, Air Force, and we hope the Air Transport Association at Albuquerque, New Mexico, from May 18 to May 20, 1992. A second theme will be



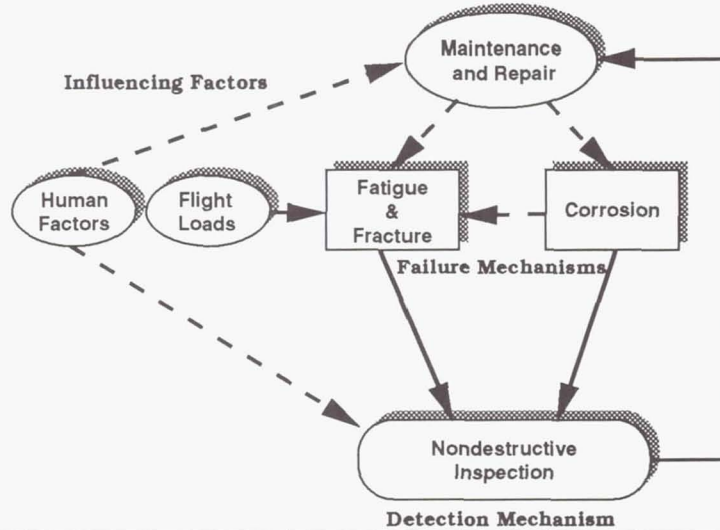
encouraging airframe and engine manufacturers, transport and commuter air carriers, repair and maintenance stations, and any other potential users, to tell us what their respective needs and perceptions are for improved NDI.

#### CONCLUSION

In the past six months, the NDI portion of the Aging Aircraft Program has been thoroughly coordinated with the Airworthiness Assurance R&D Working Group (AATF) and various FAA Regulations and Compliance Offices through a series of briefings and personal contacts. In the area of engine inspection (New England Directorate) and small airplane category problems (Central Region Directorate) in particular, there has been a major redirection of tasks within the NDI Program to accommodate their respective requirements.

The NDI Program has been structured and is managed to ensure that: meaningful issues are identified; R&D tasks are selected to address these issues; maximum use is made of technologies developed elsewhere; innovative R&D is encouraged and integrated; useful products are derived from the R&D; and R&D products are delivered to the people who can make a difference in aviation safety.

## RELATIONSHIP OF AIRCRAFT MAINTENANCE SYSTEM FACTORS

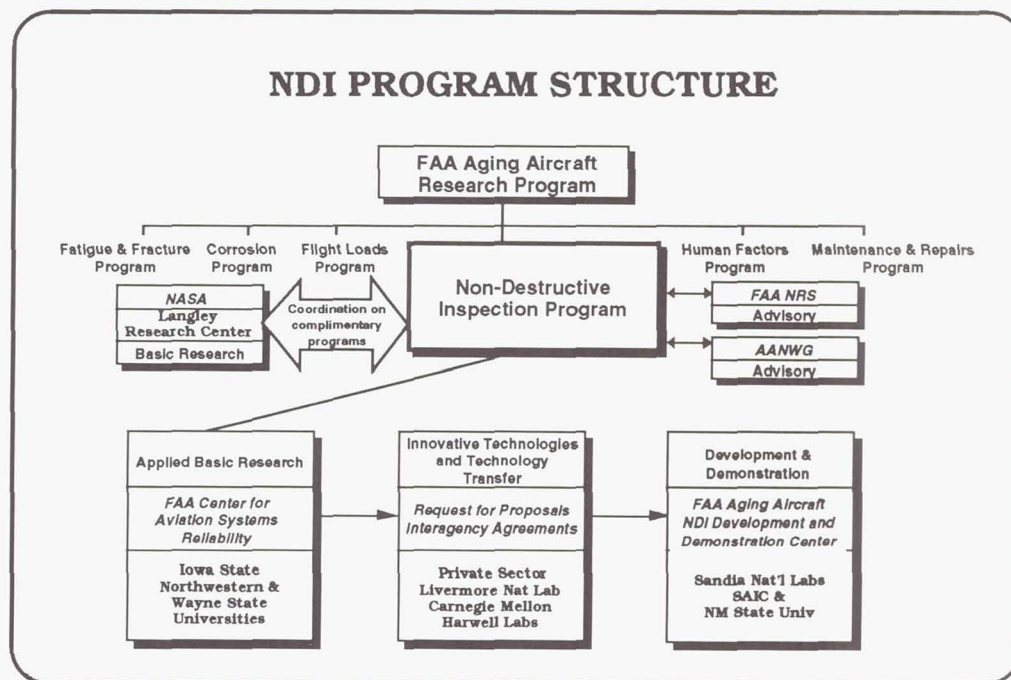


**Figure 1.**

## NONDESTRUCTIVE INSPECTION IN COMMERCIAL AVIATION

Current Techniques	Common Tasks	Shortcomings	Potential New Technologies
Visual	Airframe Cracks	Poor reliability Small cracks not seen False calls due to scratches	Broad Scan Systems Improved visual techniques Visual assistance equipment
	Corrosion	Requires good visual access	Inaccessible area NDI Neutron Radiography Corrosion monitoring
Eddy Current	Airframe Cracks	Hard to interpret signals Small cracks near fasteners hard to "see" Time consuming for large areas Surface must be accessible Metals only	Auto signal interpretation Signal processing, advanced ultrasound Broad Scan Systems
	Corrosion (thickness measurement)	Thinning only - early phases undetected Requires access False calls due to thinning	Neutron Radiography Corrosion monitoring
Ultrasound	Disbond (void) detection	Many disbands, e.g. lap joint "kissing disbands", have no voids, are not detected. Rqs skilled set up & interpretation	Laser Holography Infrared Thermography Advanced Ultrasound
	Corrosion (thickness measurement)	Thinning only early phases undetected Rqs surface access Rqs skilled set up & interpretation False calls due to thinning	Neutron Radiography Corrosion monitoring
	Airframe Cracks	Thinning only early phases undetected Rqs surface access Rqs skilled set up & interpretation False calls due to thinning	Automation Auto signal interp Signal processing Broad Scan Systems
Radiography	Airframe Cracks ("buried")	Rqs skilled set up & interpretation Directional-geometry dependent Rqs 2 sided access Radiation hazard	CAT Scan Back Scatter Radiography
Magnetic Particle	Landing gear cracks, stress corrosion	Ferromagnetic (steel) only Surface must be accessible Directional, geometry dependent Demagnetization required	Advanced Ultrasound CAT Scan
Penetrant	Engine Cracks	Surface cracks only Reliability Time consuming	Improved Engine NDI Microsensors

**Figure 2.**



**Figure 3.**



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## INSPECTION OF AGING AIRCRAFT - A MANUFACTURER'S PERSPECTIVE

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### SUMMARY

Douglas in conjunction with operators and regulators has established interrelated programs to identify and address issues regarding inspection of aging aircraft. These inspection programs consist of: Supplemental Inspection Documents, Corrosion Prevention and Control Documents, Repair Assessment Documents, and Service Bulletin Compliance Documents. In addition, airframe manufacturers perform extended airframe fatigue tests to deal with potential problems before they can develop in the fleet. Lastly, NDI plays a role in all these programs through the detection of cracks, corrosion, and disbonds. However, improved and more cost effective NDI methods are needed. Some methods such as magneto-optic imaging, electronic shearography, Deffracto-Sight, and multi-parameter eddy current testing appear viable for near-term improvements in NDI of aging aircraft.

### INTRODUCTION

After an airplane enters service, on going inspection and maintenance of its structure are essential to ensure a continuous high level of safety. These maintenance programs are specified and approved by the certifying agencies. However, these aircraft will eventually reach an age where an increase in corrosion and fatigue cracking can be expected (Fig. 1). Industry and airworthiness experts recognize that an additional structural inspection program, which would supplement the existing operator maintenance programs, is necessary for aging aircraft.

Manufacturers, operators, and regulators have been studying the question of how to assure the structural integrity of aging aircraft since 1978. In that year the British CAA published criteria for a structural audit on aging aircraft. The FAA published similar requirements in 1981. Both of these documents directed the development of Supplemental Inspection Documents (SIDs). The SIDs are intended to extend the routine maintenance program in order to detect predicted but heretofore undetected fatigue damage in older aircraft before it became a safety problem. Accordingly, about 1983, DAC and the operators established an Industry Steering Committee (ISC) to oversee the development of SIDs for each of its aircraft models.

The SIDs identify the principal structural elements (PSEs) on each aircraft. A PSE is defined as "a structural part of assembly of parts whose failure, if it remained undetected, could lead to loss of the aircraft". SIDs also identify the inspection methods and procedures associated with each PSE. Briefly, this is an inspection program to supplement or adjust existing structural inspection programs, as required, to ensure the continued safety of older aircraft.

The primary areas of concern are aircraft that have exceeded their original design life goals, shown in Figure 1. There is a significant number of Douglas built aircraft in this category and definite steps have been taken to ensure the adequacy of the maintenance in all areas. Douglas, with the cooperation of airlines, regulators, and NDT equipment manufacturers, has established five interrelated programs to identify and address issues regarding inspection of aging aircraft. These five programs are:

1. Supplemental Inspection Documents
2. Corrosion Prevention and Control Documents
3. Aging Aircraft Repair Assessment Documents
4. Service Action Requirements Program
5. Life Extension Fatigue Tests

#### DESIGN SERVICE-LIFE GOAL

Douglas' initial design goals are set to an equivalent economic life of 20 years. To meet this requirement, Douglas designs and builds each aircraft for a minimum average useful life of 40 years. This is done by selecting design stress levels, materials, and design features that provide all requisite conditions. The airframe is designed to provide relatively crack-free structure under normal operating conditions for the service-life goal of the airplane.

In the development of any new aircraft, the design concept is evaluated in test programs that range from small details to the full-scale article. Tests are performed both to satisfy regulatory requirements and to ensure that the design goals of the structure are met. Testing begins early in the design process on structural details of portions of the aircraft. Various design concepts are evaluated for structural strength, fatigue life, and damage detectability. The evaluation process identifies the design concepts that meet the design goals. Full-scale portions of the airplane are placed in test fixtures and fatigue tested, using cycle-by-cycle and flight-by-flight loading, for at least two projected lifetimes. If premature failures are encountered during this phase of testing, the design is modified and service bulletins are issued to correct deficiencies in any delivered aircraft. At Douglas, at least one lifetime of fatigue testing is completed before an aircraft enters commercial service.

#### SUPPLEMENTAL STRUCTURAL INSPECTION PROGRAM

Douglas has developed a SID program based on fleet-leader-operator sampling. During the sampling, inspections are carried out on a PSE-by-PSE basis. Symmetrical structure results in two samples, left and right, per aircraft. For sampling purposes, one or both sides of the aircraft may be inspected. It is important to note that each PSE always stands by itself; i.e., inspection start points, inspection intervals, etc., are generally different for each PSE.

The Douglas SID inspection program is established from statistical probabilistic concepts based on a log-normal distribution of having and detecting a crack. These concepts are based on each PSE's fatigue-life



estimate, its damage tolerance characteristics, and the NDI method. The basic concepts are illustrated in Figure 2. In the sampling program, the aircraft are to be inspected before they exceed the fatigue-life threshold,  $N_{TH}$ , with inspections starting at  $N_{TH}/2$ .

After a PSE exceeds  $N_{TH}$ , the interval between inspections is a function of the NDI method used and the crack growth and residual strength characteristics of the PSE. The interval is set to equal  $\Delta N_{di}$  (Figure 2) divided by 2.  $\Delta N_{di}$  is the time for a crack to grow from a discoverable length,  $a_{det}$ , to the instability length,  $a_{inst}$ , which is the crack length that would produce instability failure due to limit load application.  $\Delta N_{di}$  is divided by 2 in order to provide two opportunities to detect the crack after it has reached a detectable length and before it reaches the instability length.

The Douglas SIDs were completed and released as follows:

- o DC-8, January 1986
- o DC-9, June 1986
- o DC-3, May 1988 (after 50 years of service)
- o DC-10, January 1989
- o DC-6, March 1991
- o The DC-7 and MD-80 SIDs are presently being developed.

#### CRACKS VS NOTCHES

For eddy current inspection, simulated structure is fabricated with electrical discharge machining (EDM) notches of different sizes. The notched specimens are then used to develop preliminary procedures and determine the detectable flaw size as it applies to each PSE. The detectable flaw size is assumed to be equivalent to the size of the EDM notch used in setting up the instrument. However, some questions were raised concerning the validity of this procedure. To answer these concerns, both Douglas and Boeing engineers performed a series of eddy current tests where they compared the responses from fatigue cracks and EDM notches in aluminum specimens. The results in Figure 3 showed good agreement between cracks and notches of similar sizes in aluminum specimens.

#### CORROSION PREVENTION AND CONTROL DOCUMENTS

Corrosion is generally considered a bigger maintenance problem than fatigue cracking. Hence, the original equipment manufacturers (OEMs) prepared Corrosion Prevention and Control Documents for each model aircraft. These documents are used to establish a Corrosion Prevention and Control Program for older (by age) aircraft. Operators are required to apply a CP and CP on all aircraft that reach or exceed the Baseline Program Implementation Age for each corrosion task. The responsibility to establish, maintain, monitor, and regulate the program within the guidelines lies primarily between the operators and their regulatory agencies, with the OEM participating as technical advisor when requested. The program is intended to inspect all primary structures as well as all items that are susceptible to corrosion in a given zone. Each aircraft is broken down into zones with specific inspection tasks within each

zone. Visual inspection is the primary detection with NDI being used for detection in non-accessible areas.

Each operator is also given a copy of a Corrosion Control Handbook (MDC K0805). This handbook describes methods for detection, removal, and prevention of corrosion. The handbook contains a rather large section on the use of NDI methods for detection of various forms of corrosion (Figure 4).

Corrosion assessment is broken down into three "Levels":

- Level 1. Is local and can be re-worked within limits.  
Is local and exceeds limits but caused by non-typical event.  
Light corrosion between inspections but now exceeds limits due to cumulative blend-out and requires repair.
- Level 2. Occurs between inspections; exceeds limits, requires repair.  
Widespread between inspections and requires blend-outs less than limits.
- Level 3. Exceeds limits; potential airworthiness concern.

The Corrosion Prevention and Control Documents for DC-8, DC-9/MD-80, and DC-10 were released to operators in the Spring of 1990. In addition to the general guidelines, which are the same for each model, the specific documents identify Service Bulletins and chapters in the Maintenance Manual and Structural Repair Manual concerning corrosion issues for that model aircraft.

#### REPAIR ASSESSMENT DOCUMENTS

Numerous repairs have been made on aircraft structure. The majority of these repairs meet the minimum standards for strength but some do not meet the current standards for resistance to fatigue and corrosion.

The concern that existing repairs should be reviewed for damage tolerance requirements and may need special inspections/replacement to ensure structural integrity resulted in efforts by Model Task Groups (MTGs) to develop an operator usable system to evaluate repairs on aircraft. Because of the enormous scope of this effort it was decided that guidance material be developed which would cover the review of a majority of existing aircraft repairs. Repair assessment guidelines have been prepared to cover each airplane model and are scheduled to be completed in phases. The Model Task Groups have been working on the guidelines for Phase I (Fuselage Pressure Shell) for about two years. Final completion for Phase I is expected some time in 1992.

In addition, an industry-operator task force is in the process of making recommendations to the FAA on the subject of repairs and the damage tolerance analysis required to certify the repair for a given life. Once completed, this information will be provided in the model specific Structural Repair Manual (SRM).



The Primary objectives of the Repair Assessment Program are:

1. To provide a practical methodology in the form of guidance material (based on damage tolerance principles) that will allow a majority of existing repairs to be evaluated by operators without complex analysis.
2. To direct the repair evaluation process towards showing that an installed repair which replaces the strength and durability of the original structure is found damage tolerant if the existing inspection levels are adequate.
3. To develop a repair evaluation process such that it will identify when and what supplemental inspections (threshold, method, interval) are required.

Nondestructive inspections may be required in the repair assessment process to identify the following concerns:

- o Is the repair or surrounding structure damaged?
- o Are the fasteners less than or equal to 1/8 inch diameter?
- o What is the thickness of the repair?
- o Is the repair material, Al, Ti, or stainless steel?

The NDI techniques for answering the above questions are contained in the model specific SRMs. Basically, the repair materials are identifiable using eddy current conductivity testing. Plating thickness is determined by using ultrasonic thickness testing. Fastener diameters are identified by either removing one of the fasteners or by X-ray radiographic inspection. It is very difficult to determine the difference in shank diameter for 1/8 and 3/32 inch fasteners because their head diameters are almost equal.

Four categories of repairs have been developed to define structural maintenance requirements:

- |              |  |
|--------------|--|
| Category - A | Satisfy damage tolerance requirements. No inspection required.   |
| Category - B | Meet design certification but must be periodically inspected to satisfy damage tolerance requirements. |
| Category - C | Temporary nature requiring periodic inspection and replacement at a certain time limit.                |
| Category - D | Do not meet design requirements and must be upgraded or replaced before further flight.                |

It has been shown that repairs can degrade the overall fatigue quality of the basic structure. It has also been shown that the most critical locations for future fatigue damage are in the basic skin at the first attachment row in the doubler. Because the doubler covers a possible crack in the basic skin, visual inspection becomes impossible until the crack grows past the edges of the doubler. Hence, Douglas engineers developed ultrasonic shear wave inspection to detect skin cracks under the doubler and low frequency eddy current techniques to detect skin cracks through the doubler. These procedures are contained in the Structural Repair Manuals for use by operators.

In the present program, all repairs in the fuselage pressure vessel will be evaluated using a Repair Categorization Questionnaire.



## SERVICE ACTION REQUIREMENTS PROGRAM

In this program, the Model Task Groups (MTGs) evaluated all structure-related service bulletins to determine which service actions should be recommended for termination from special repetitive inspections and to determine the thresholds for incorporation of the structural modification which terminates the special inspections.

The DC-8, DC-9, and DC-10 MTGs reviewed more than 4,700 structure-related service bulletins dated through December 1988. This phase of the program was completed between October 1988 and December of 1989. A total of 159 of the 4,700 service actions were selected for incorporation by the terminating action, which eliminated the repetitive inspection process. It is expected that closing action for these service items will be required within 4 years of the effective date of the airworthiness directive (AD) or by the prescribed "closeout" life, whichever occurs later. The closeout life established for each service action is either the test supported life or the time when there is a significant possibility of missing a crack through inspection.

Current task group activity is concentrating on structure-related service bulletins released between January 1989 and November 1990.

A nondestructive test engineer was a member of each task group who provided improved or updated NDI procedures to those bulletins deemed necessary to obtain a more reliable inspection.

## EXTENDED AIRFRAME FATIGUE TESTING

In the original design of an aircraft, the usage of the aircraft is estimated on a 20-year economic life and a specific utilization. These estimates are based on the structural design requirements and the customer's anticipated use of the aircraft. The original fatigue goals based on these initial estimates and the fatigue test that was conducted to verify the design may at some point not adequately shelter the aircraft in the fleet. At this point a new fatigue test of the structure is appropriate to characterize the fatigue behavior of the aircraft. The new fatigue test, as it surpasses the original test life, will reveal new problem areas that have not developed in service. This allows the manufacturer to deal with potential problems before they can develop in the fleet.

In 1981, Douglas purchased DC-9 No. 3, a 15-year-old aircraft which had accumulated more than 66,500 landings. The aircraft was stripped of all interior items and the structure was thoroughly inspected. The structure was found to be remarkably free of fatigue cracks and corrosion. Defective areas were either repaired or marked for evaluation during the test.

The fuselage shell was pressure-cycled to a total of 208,000 flights, including flights in service as well as in test. Selected areas of the structure were periodically evaluated by NDI techniques during the test. Upon completion of the test, an extensive detailed tear-down inspection of the fuselage structure was conducted. More than 8,000 fastener holes were checked for cracks in 22 feet of fuselage lap splices. The specimens were cleaned,

etched, and visually inspected at 5X magnification. Some 6,000 attachments were removed from the right wing and inspected using eddy current bolt-hole techniques. Then, coupons were removed from the wing and empennage and fatigue tested. Results of the test showed that the DC-9 structure was capable of achieving a life of 104,000 flights, free from widespread fatigue cracking.

As part of the MD-80 certification, the pylon and aft fuselage structure was tested to 3 lifetimes (150,000 flights), and the aft pressure bulkhead was tested to more than three lifetimes (183,000 flights). Numerous radiographic and eddy current inspections were conducted during the aft pressure bulkhead fatigue test. More recently, the DC-10 horizontal tail assembly underwent a spectrum loaded fatigue test to validate fatigue preventive modifications in the rear spar area.

## NDI RESEARCH AND DEVELOPMENT

The FAA's National Aging Aircraft Research Program Plan describes planned research efforts designed to enhance aircraft safety through a better understanding of airframe structural performance and maintainability. The program includes studies on nondestructive inspection (NDI) methods, techniques, and practices with emphasis on reduction of inspection costs without affecting reliability. The NDI program will focus on detection of fatigue cracks, debonding, and corrosion.

Improvements in NDI technology continue through the cooperative efforts of researchers, engineers, and technicians. To that end, the engineers at Douglas have looked for emerging NDI technology which could be used almost immediately for on-aircraft inspections. Following is a description of five promising NDI methods:

### Low-Frequency Eddy Current Testing

Nondestructive Inspection (NDI) programs consistently show that eddy current inspection is very reliable in detecting small, tight fatigue cracks. Because of these findings, eddy current testing became the primary crack detection method used in Douglas aging aircraft inspection programs (ref. 1). High frequency (above 10 kHz) eddy current testing is used to detect surface cracks. Low frequency (100 Hz to 10 kHz) eddy current testing is used to detect subsurface cracks in aluminum aircraft structure. The lower the test frequency, the deeper the eddy currents propagate in the structure.

Douglas NDT engineers, in conjunction with equipment manufacturers, have been developing the low-frequency eddy current technology for the past decade. Most of the developments were in the areas of improved portable impedance plane instruments that operate at low frequencies, driver/receiver probes, and applications and limitations of this technology.

In-service inspection techniques are available for detecting cracks in aircraft using a driver/receiver sliding-probe eddy current system for improved hole inspection, without removing the fasteners. Better crack sensitivity and reliability are obtained when using the sliding-probe system (see Figure 5).



The author is presently evaluating multi-parameter eddy current to improve the signal-to-noise response obtained in many practical applications. Multi-parameter eddy current testing expands the capabilities of the eddy current method. Today's instrumentation is capable of operating at two or more test frequencies, displaying multiple channels of data, exciting more than one coil configuration, and combining or mixing raw data channels to generate new data channels. These tools of multi-parameter testing can provide more information about the test piece as well as decreased inspection time.

#### Magneto-Optic Eddy Current Imager

PRI Instrumentation has recently developed a new inspection technology called "magneto-optic/eddy current imaging" (ref. 2). This technology makes it possible to generate real-time images of defects, such as fatigue cracks and corrosion. An imaging system based on magneto-optic principles has been developed that is capable of inducing eddy currents in the frequency range from 1.6 to 100 kHz. At the higher frequencies, the instrument can image and detect small, tight cracks near rivets on the outer surface of aluminum aircraft skins (Figure 6). At lower frequencies, the instrument can detect an image of second layer cracking and some types of corrosion in aluminum.

The advantages of this new visually-based technique include increased inspection speed, more intuitive and easily interpreted inspection information, elimination of false calls, elimination of the need for paint removal, and the ability to document results with videotape of 35mm cameras.

The author witnessed the development of the MOI (by Gerald Fitzpatrick) for the past few years and provided cracked samples for evaluation of the prototype unit. Douglas is presently approving the MOI for use in surface crack detection in conjunction with DC-10 SID inspections at American Airlines.

#### Electronic Shearography

Shearography brings the sensitivity of holographic interferometry to the detection of skin-to-core unbonds and deep unbonds in both honeycomb and foam sandwich structures during fabrication inspection. Similar structures can be inspected for disbonds caused by in-service stresses.

Unlike holographic interferometry, shearography is a real-time technique that is all electronic and insensitive to environmental vibration and ambient light. As with holography, an appropriate stressing technique is required to allow a sub-surface flaw to induce a strain concentration on the surface of the part. This may be achieved using vacuum, thermal, or pressure. To detect water in honeycomb structures, microwave excitation is used as a stressing technique. For composite structures, the primary stressing techniques are thermal loading for detecting delamination or impact damage and pressure reduction (vacuum) loading for detecting unbonds/disbonds. Disbonds in fuselage lap joints are easily detected by pressurizing the aircraft.

Electronic shearography was invented by Professor Y. Y. Hung at Oakland University in 1982 and licensed exclusively to Laser Technology, Inc. Shearography uses a laser to illuminate an area on the test part for inspection. Figure 7 shows a schematic diagram of a typical shearography system. The laser beam is introduced to the test part through a fiber-optic cable. The output is an image-processed video display in color, which can show both size and shape of subsurface defects as well as qualitative measures of depth.



John Tyson of Laser Technology, Inc., reported successful detection of corrosion, disbands and cracks in aluminum aircraft structure (ref. 3). Further studies are being conducted by LTI under funding by the FAA. The author continues to follow the successful results obtained by LTI and is looking for applications for this technology in both fabrication inspection of composites and in-service inspection of aircraft structure for disbands, corrosion, and cracks.

#### DiffRACTO-Sight or D-Sight

DiffRACTO-Sight or D-Sight (developed by DiffRACTO Ltd. of Windsor, Ontario, Canada) is a patented method of visualizing surface distortions, depressions, or protrusions as small as  $10\mu\text{m}$ . It is a real-time technique that is particularly applicable to rapid inspection of large surfaces. Komorowski of the National Aeronautical Establishment of the National Research Council of Canada suggested using this technique to inspect composite structures for "barely visible impact damage" (BVID). Computer-based image processing has been applied to D-Sight images. An image from a previous inspection can be directly compared to current results for quick identification of areas where changes in surface features have occurred.

The optical setup for D-Sight consists of a light source, a retroreflective screen, and the object being inspected (Figure 8). The surface being inspected must be reflective. Both flat and moderately curved surfaces can be inspected using this method (ref. 4).

The D-Sight effect can be explained using geometric optical principles. If a flat surface with an indentation is inspected, the light striking the indentation is deflected. It then strikes the retroreflective screen at a point removed from the light rays reflected from the area surrounding the indentation. The retroreflective screen attempts to return all these rays to the points on the inspected surface from which they were first reflected. However, the screen, consisting of numerous glass beads, returns a cone of light to the surface. This imperfection of the retroreflective screen creates the D-Sight effect. By backlighting the defect, the technique increases the light intensity on one side of the indentation and reduces it on the opposite side.

Douglas currently has a D-Sight setup and is evaluating a variety of flawed specimens to determine applications for both fabrication and in-service inspections. We are also working with Omar Hageniers of DiffRACTO in helping to advance this exciting technology. Presently, this technique has proven successful for detection of surface abnormalities such as impact damage, waviness, buckles, dents, and flaws such as corrosion bulging, and cracks.

#### Computerized Ultrasonic Test Simulation (CUTS)

Ultrasonic shear-wave (angle beam) inspection has been used routinely for detection of subsurface cracks in aircraft structure especially adjacent to fasteners. To develop an in-service ultrasonic inspection technique, the NDT engineer has an actual size cross-sectional view of the aircraft structure with the crack position accurately located. Working backwards from the crack position, he determines the required angle of refracted sound and the position of the search unit on the external surface of the test part. This technique development exercise requires proper calculation of the refracted wave and considerable time in making the drawings. To improve this situation, Douglas

engineers programmed a personal computer with Snell's equations for reflection and refraction of sound in a number of materials having different sound propagation velocities.

To insure that a particular part will be properly inspected, the path of the ultrasonic beam within the part must be known. However, when a beam impinges upon the surface of a part it refracts and reflects within the part. For parts of complex geometry, it is often difficult and time consuming to calculate the paths of these refracted and reflected beams. For each location and incident angle of the transducer, calculations are made to determine if the refraction of the incident beam will intersect the area of interest within the specimen or part. Manual calculations can take considerable time, but the CUTS software performs these calculations in real time. Therefore, as the user repositions the transducer he/she instantly sees the results in real time of the new transducer position and/or angle.

The CUTS software models wave behavior in parts made of isotropic, homogenous materials. The software was written utilizing FORTRAN and "C" programming languages. First, the user inputs the name of the file containing the part geometry, then he/she selects the material that the part is made from. The program draws the part on the screen and places an icon representing the ultrasonic transducer (search unit) near the part at an initial incident angle of 45 degrees. This incident angle is set with a velocity of sound in acrylic (for contact testing) and a velocity in water (for immersion testing). Transducer incident angles are measured with respect to the negative y-axis. Now the user is able to move the transducer icon around the part and observe the effects of the different incident angles and locations in real time. (See Figure 9).

The CUTS program has been useful for developing ultrasonic techniques for in-service inspections especially on parts with complex geometries. It has also been useful for determining where unwanted reflections are generated within the part.

#### CONCLUDING REMARKS

The purpose of the FAA's Aging Aircraft Research Program is to assure that aircraft are adequately inspected and maintained in an airworthy condition as they are operated up to and beyond their original design life objectives. Douglas, in conjunction with operators, manufacturers, and regulators continues to develop and improve criteria and control programs for protection of the aging aircraft fleet. This protection is provided by the mandated SID, Corrosion Assessment, Repair Assessment, and Service Bulletin compliance programs. In addition, the fleet is also protected by the manufacturers' extended airframe fatigue testing programs and the industries' improvements in NDI technology.

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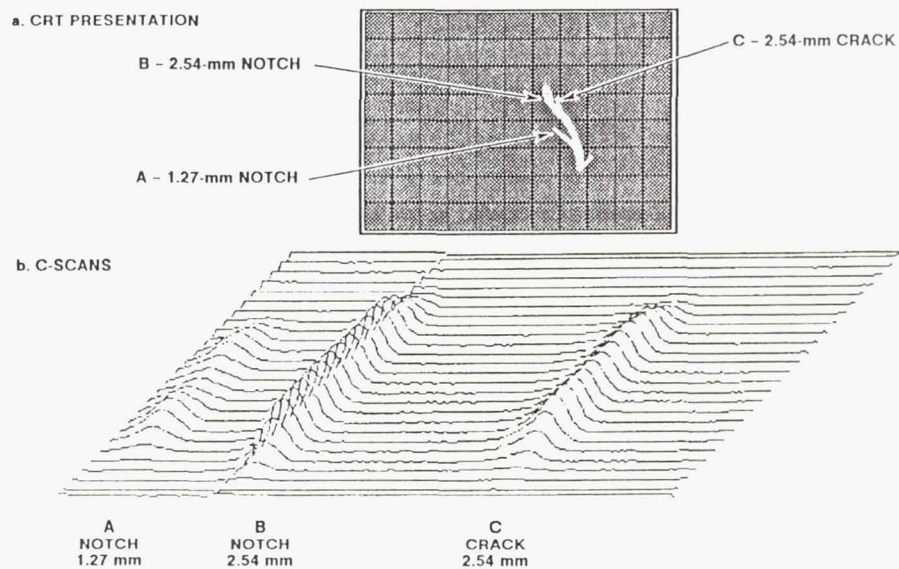







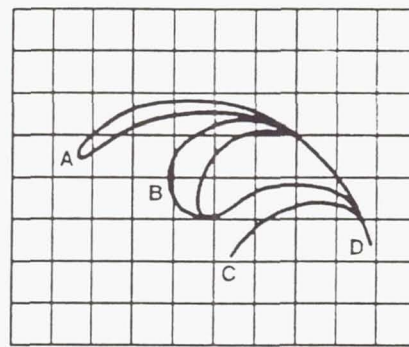


Figure 3. Eddy Current Results for Small Notches and Crack

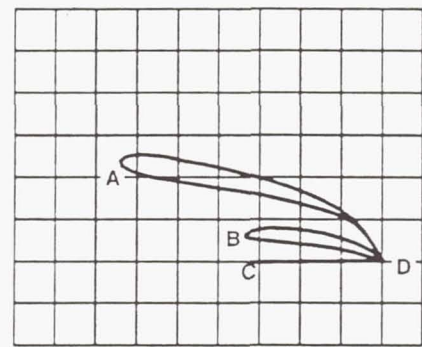
DETECTION METHODS	EQUIPMENT					
	SIZE	MOBILITY	AUTOMATED	SPEED	COST	AVAILABILITY
VISUAL 	SMALL	GOOD	NO	FAST	LOW	YES
TAP TEST 	SMALL	GOOD	POSSIBLE	FAST	LOW	YES
ULTRASONIC 	MEDIUM	GOOD	POSSIBLE	MODERATE	MODERATE	YES
EDDY CURRENT 	SMALL TO MEDIUM	GOOD	POSSIBLE	MODERATE	MODERATE	YES
X RAY RADIOGRAPHY 	MEDIUM TO LARGE	FAIR	NO	SLOW	HIGH	MOST SHOPS
NEUTRON RADIOGRAPHY 	LARGE	POOR	NO	SLOW	VERY HIGH	RARE
*ACOUSTIC EMISSION WITH HEAT 	MEDIUM	FAIR	NO	MODERATE	MODERATE	VERY FEW SHOPS

\*PARTS MUST BE REMOVED FROM AIRCRAFT FOR TEST.

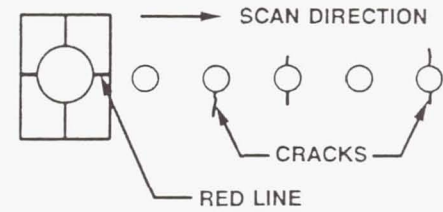
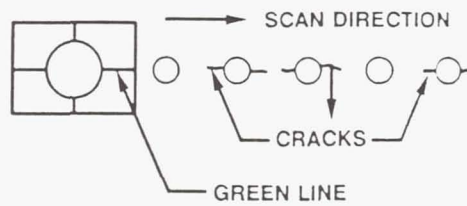
Figure 4. Rating of Test Methods



(a) NORMAL SCANNING



(b) 90-DEGREES SCANNING

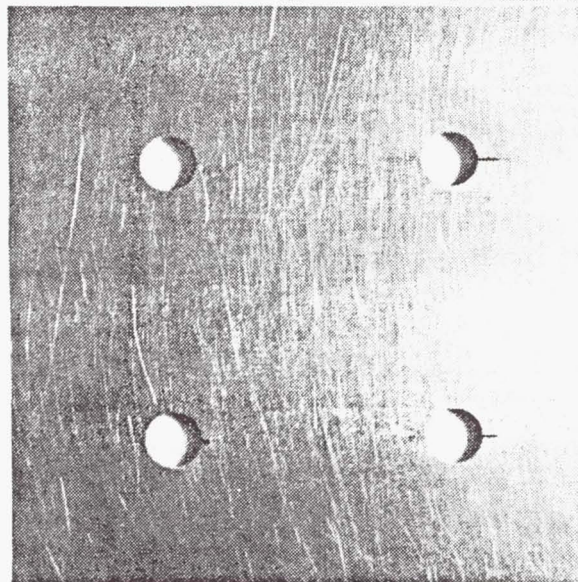
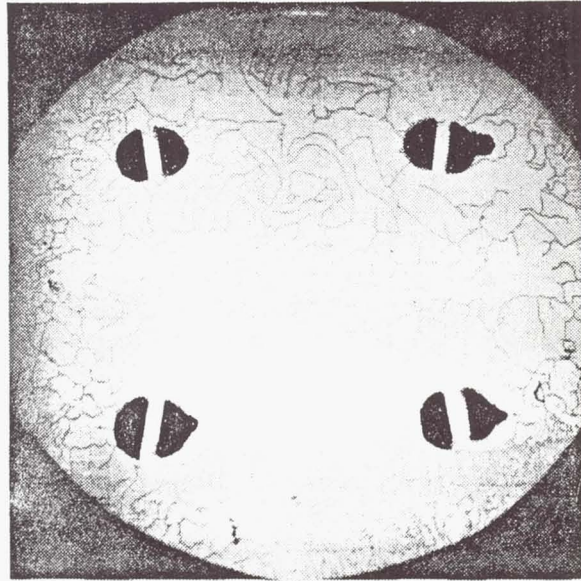


- |   |  |
|---|--|
| A. CRACK AT BOTH SIDES OF FASTENER HOLE | C. UNCRACKED FASTENER HOLE                   |
| B. CRACK AT ONE SIDE OF FASTENER HOLE   | D. ALUMINUM RESPONSE AWAY FROM FASTENER HOLE |

Figure 5. Sliding Probe System for Fastener Hole Crack Detection

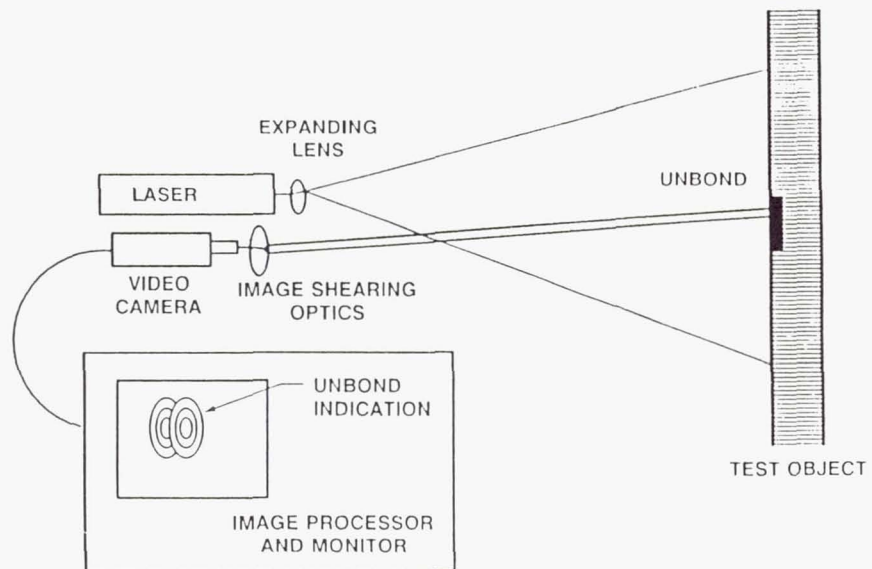


(a)



(b)

Figure 6. (a) Magneto-optic eddy current image. (b) Photo of the test piece. This 0.06" thick aluminum panel contains four 0.25 inch diameter holes with 0.020, 0.040, 0.060, and 0.080 inch long EDM notches emanating from them. The order is counterclockwise beginning with the 0.02" long EDM notch in the upper left hand corner of (a) and (b).



COURTESY OF LASER TECHNOLOGY, INC.

Figure 7. Schematic Diagram of Electronic Shearography

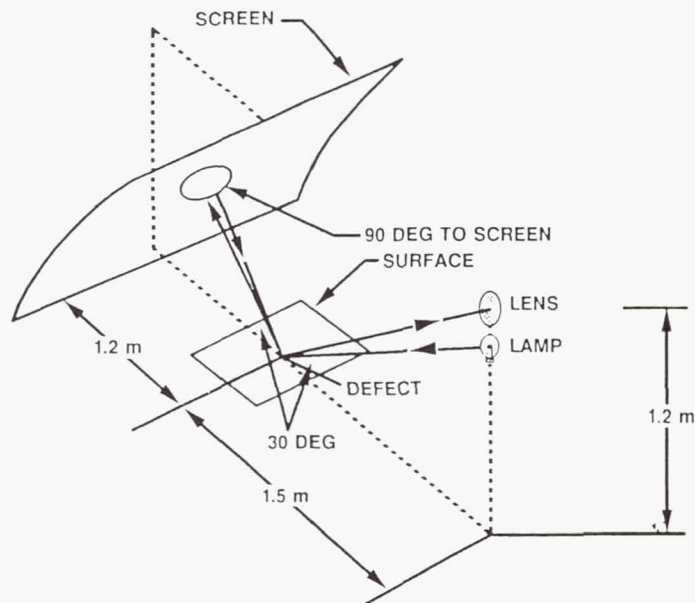


Figure 8. Diffracto-Sight Optical Setup

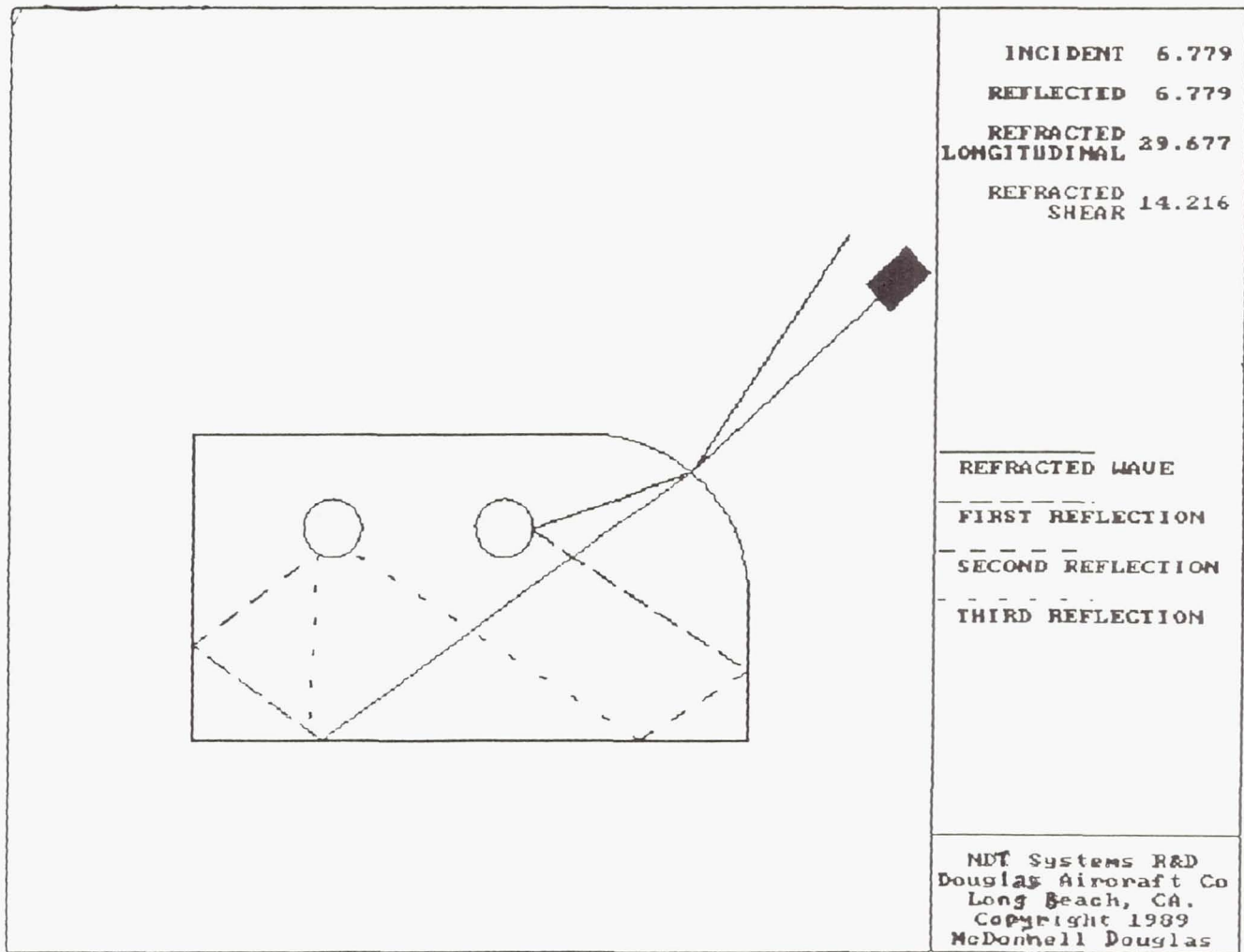


Figure 9. Computerized Ultrasonic Test Simulation (CUTS)



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# THERMAL QNDE DETECTION OF AIRFRAME DISBONDS

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## SUMMARY

Thermographic characterization of aircraft bonded lap joints offers a quick noncontacting technique to acquire information for structural integrity assessment. This paper discusses recent research to optimize the technique and determine the limits of its applicability. The temperature of the outer surface of the lap joint is increased by the application of heat flux from either flash or quartz lamp heaters. The time dependence of the surface temperature of the lap joint is imaged radiometrically. Measurements are presented for a range of specimens, ranging from samples fabricated with well characterized disbonds to actual aircraft. A technique for processing these images to enhance the contrast between bonded and disbonded regions of the lap joint is presented. Numerical models of the technique simulate the procedure. These simulations provide a cost efficient method for optimizing the technique by varying parameters such as the time for application of heat. These simulations facilitate the definition of parameters difficult to determine experimentally, such as the minimum air gap required for a disbond to be detected. Good agreement between measurements and these simulations is found.

## INTRODUCTION

With the increased age of the commercial aircraft fleet, there exists a greater need for the development of new NDE techniques for the detection of critical flaws in aircraft airframes. Current techniques are considered either too time consuming or unreliable as a primary basis for integrity resulting in a requirement for major mandatory modification to the existing fleet. Improved NDE techniques offer the possibility for improved safety and reliability at reduced cost.

A major thrust of this effort is the development of large area scanning techniques. Current inspection techniques are unaided visual inspection or point measurements using a small hand held probe. Eddy current detection of cracks at rivets and sonic bond testers are characteristic of current technology being applied in maintenance facilities. These inspections are manpower intensive, requiring teams of inspectors several days to inspect the entire aircraft. Inspection reliability is reduced by human factors such as fatigue and boredom. These inspections are further compromised by the correlative basis for the techniques. Based

primarily on testing samples with fabricated defects, the ultimate reliability of the technique depends on proper fabrication of standards with defects which accurately represent in-service airframe defects.

In contrast to these techniques, much current research is focused on large area inspection techniques. A few of the areas currently under investigation are thermal imaging, acoustic emission, scanning array ultrasonics, coherent optics, radiography, magnetic field visualization and visual enhancement. Research is being pursued to develop a science base for these techniques to enable quantitative characterization of the aircraft structure. A quantitative characterization permits computer enhancement for highlighting areas of concern and reducing human factors such as fatigue and boredom. Large area scans reduce inspection time, reduce manpower cost and out of service time for the aircraft. Finally a less costly inspection improves the safety of the aircraft by economically allowing more frequent inspection.

Typical of the new large area inspection techniques is infrared thermography. Thermographic imaging does not require physical contact between the inspection system and the aircraft. The prediction of the thermal response of an aircraft has a well established science and engineering base from parallel studies to determine the thermal response of aerospace structures to given thermal loads. Infrared thermography is performed with an infrared imager capable of scanning large areas in a fraction of a second. Recent technological developments in digitized thermography have significantly increased the number of successful applications. In particular, advances in inexpensive image processors have increased the signal to noise ratio for thermographic images and enabled advances in post processing procedures. New staring array infrared imagers also have increased signal to noise relative to a single scanned detector. Thermographic imaging has an additional advantage of not requiring physical coupling between the inspection system and the aircraft.

Previous studies have shown the feasibility of thermographic techniques for detection of disbonds in bonded structures (ref. 1-7). Presented here is an overview of an effort at NASA Langley Research Center to apply thermographic techniques to the detection of disbonds in airframe lap joints. This involves computer simulation to develop a better understanding of the thermal processes particular to airframe lap joints and determine some of the limitations of the technique. Measurements are also performed on samples with both fabricated defects as well as samples removed from aircraft. Finally the technique is tested both on a NASA Langley 737 and aircraft at maintenance facilities.



## COMPUTER SIMULATION OF TECHNIQUE

The thermographic technique consists of the application of heat to the surface of a structure and subsequent measurement of the surface temperatures as a function of time. To inspect a lap joint on an aircraft, the heat is applied to the exterior of the aircraft and the exterior skin temperature is measured. By heating the sample, a temperature differential is created between the lap joint regions and regions adjacent to it due to the larger heat capacity of the lap joint. For disbonded regions of the lap joint, the heat flow from the upper to lower layer of the lap joint is reduced. This reduction of heat flow is reflected in an increase in temperature over the disbonded region relative to the bonded regions.

Computer simulations of the thermographic technique were performed to help establish a better technical base for this approach. Computer simulations are a very useful tool for optimization and then determining the limitations of the technique. It is a cost effective method for considering a large variety of structures and defects. Optimization of the technique is easier since exact control can be exercised over the different parameters of interest. Examination of the sensitivity of the technique to defects which are costly and difficult to fabricate can be simulated with relative ease.

A two and three dimensional time dependent finite element heat transfer algorithm developed at Lawrence Livermore National Laboratory is used to solve the time dependence of the temperature of the structure (ref. 8). Research indicated adequate simulations require inclusion of the heat flow in the aluminum with the adhesive layer represented as a contact resistance. A disbond is simulated by increasing the contact resistance by an amount proportional to the width of the air gap forming the disbond. Details of this work are given in reference 9.

Typically the lap joint and 15.0 cm to either side of the 7.6 cm wide lap joint are included in the simulation. Since the aluminum is only 0.1 cm thick, finite element models of this structure have difficulty with large aspect ratio cells. Recent work (ref. 10) has shown this aspect ratio difficulty can be overcome by assuming a quasistatic flow condition in the aluminum, reducing the dimension of the governing partial differential equation by one. This reduced partial differential equation no longer has a large aspect ratio in the finite elements. The reduction of the dimensionality also reduces the computational time required for the simulations.

Typical results of the simulations are shown in figure 1. Initially the temperature between the bonded and disbonded regions grows as a function of time. As the temperature differential increases, a lateral heat flow develops between the bonded and disbonded regions. When the lateral heat

flow is equivalent to the applied heat flux, the temperature differential ceases to grow. This lateral heat flow reaches an equilibrium in a period of time dependent on the shape of the disbond, with smaller disbonds obtaining an equilibrium sooner than the larger disbonds. Therefore, the optimum heating protocol depends on the size and shape of the disbond. The optimum heating time for a 2.5 cm wide disbond is approximately 8 seconds.

The simulations indicate an improvement in contrast can be obtained by taking the time derivation of the thermal images. This can be seen from the time derivative for the simulation shown in figure 1. For this case the time derivative clearly shows the disbonds, in contrast to the temperature profiles the disbonds are difficult to delineate.

The advantage of simulations is clear when determining the effect of air gap width on the detectability of a disbond. For a disbond to be detectable, the air gap must be wide enough to significantly reduce the heat flow between the lap joint adherents. Fabrication of samples with varying air gaps is very difficult and expensive. However to vary the width of the air gap in the simulation is relatively easy. To determine the effect of the width of the air gap on the contrast between bonded and disbonded cases, simulations were performed keeping the size of the disbond constant. These simulations indicate a 10  $\mu\text{m}$  air gap is required for significant contrast between bonded and disbonded regions.

#### INFRARED BOND INSPECTION SYSTEM

A schematic of the thermal measurement system is shown in figure 2. The infrared imager consists of a single liquid nitrogen cooled HgCdTe detector (8 - 12  $\mu\text{m}$ ). The single detector is scanned over the field of view to measure the infrared emission from the surface of the aircraft. Assuming the emissivity of the surface is 1, this signal is proportional to the temperature of the surface. If the emissivity of the surface is less than 1, the signal is a combination of the surface temperature and a reflection of background infrared images. An aluminum surface has an emissivity of less than 0.1; therefore infrared measurements of an aluminum surface yield images of the background infrared fields and not the surface temperature of the aluminum.

To overcome this difficulty, the aluminum is often coated with a thin layer of high emissivity material. Paint is typically an excellent emissivity coating, with a high emissivity regardless of the color in the 8-12  $\mu\text{m}$  region. Aircraft which are painted therefore require no special treatment before inspection with the thermal inspection system. For unpainted aircraft, water washable coatings are commercially available for increasing the emissivity of the surface. The work reported here was on aircraft with either a painted surface or with a self adhering sheet applied to the aircraft. This sheet has a high



emissivity and measurements show its thermal properties as a coating were only slightly worse than those of paint.

The imager converts the infrared radiation (thermal response) from the surface of the sample to a video signal which is digitized by an image processor that performed a real time average of the digitized images. The real time averaging of these images significantly increased the signal to noise ratio of the data. Averaged images were obtained for given time series determined by the microcomputer controller. The current system allows for both immediate reduction of the data and storage of the data for further analysis.

An important feature of the system for quantitative measurements is the microcomputer which controls the data acquisition and also controls the application of heat. This enables synchronization between the heating and data acquisition. Without this synchronization it is often difficult to interpret the images obtained and impossible to obtain quantitative results. Two banks of lamps were used to heat the surface of the aircraft. Tubular quartz lamps (4000 watts) with parabolic back reflectors focused the energy onto the aircraft. These imparted  $\approx 0.1$  watts/cm<sup>2</sup> to the aircraft of interest. Flash tubes with parabolic back reflectors were used as well for heat sources. They imparted  $\approx 0.2$  joules/cm<sup>2</sup> to the aircraft in less than 1/100 of a second. For all cases considered, the temperature of the aircraft was never raised more than 10° C above ambient conditions.

#### MEASUREMENT RESULTS

Initial measurements were performed on laboratory standards with fabricated disbonds. The laboratory standards used, as shown in figure 3, consisted of two sheets of aluminum bonded with a three inch overlap using a room temperature cure epoxy. The dimensions of the sheets were 61 cm by 122 cm and 0.102 cm thick. Disbonds were created by inserting pull tabs (.013 cm thick) of different dimensions into the bonded region before curing. After curing, these pull tabs were then removed to leave voids in the bond.

The data obtained on these specimens was used to compare a variety of different data acquisition and reduction techniques. A cyclic heating and data acquisition protocol was determined to be most effective for detection of disbonds (ref. 11). The data was reduced by a variety of techniques such as time derivative (ref. 12) and neural networks. The neural network gave good contrast between bonded and disbonded regions; however, this requires a training of the network on either simulations of the structure or structures which are known to have disbonds. For either case, a prior knowledge of the thermal response of the structure is required. The time derivative does not require this prior knowledge, however, it is slightly less accurate in separating bonded and disbonded



regions of the lap joint.

Typical results for a laboratory standard is shown in figure 4. All of the disbonds, the smallest being 2.5 cm wide, are discernible in these images. The visual inspection of these images suggests that the contrast between the disbonded and bonded regions is greater for the 9.6 and 18.1 second heating cycles than for the 5.3 and 35.2 second cycles.

Comparison of 9.6 and 18.1 second periods suggests that the contrast for the largest disbond continues to increase. However, for the 2.5 cm disbond it is questionable if the contrast is improving or degrading. A quantitative comparison shows the 9.6 second heating period is the optimum for the 2.5 cm disbond. However, for a 5 cm disbond the optimum period increases to 18.1 seconds. This variation of optimum heating time was predicted by the simulations of the technique.

Data typical of a well bonded portion of an aircraft are shown in figure 5 and 6. These images were obtained on an aircraft at a maintenance facility. In these images the bonded substructure of the airframe is clearly seen as the darker portion of the image. Outlined on these images is the substructure as determined from engineering schematics of the airframe. The bonded lap joint, tear straps and doublers above the window are all clearly seen in these images.

Images from regions with disbonded tear straps are shown in figure 7 and 8. Again outlined on these images is the extent of substructure as determined from engineering schematics of the airframe. In these images, the disbonded portions of substructure of the airframe are clearly detectable. After obtaining these images, the existence of the disbonds was verified by ultrasonic inspection and visual inspection of the interior structure.

#### CONCLUSIONS

Recent advances in nondestructive evaluation systems offer the possibility of increased reliability and safety at reduced cost. In particular, advances in large area scanning techniques offer significant advantages over current visual and point measurement techniques.

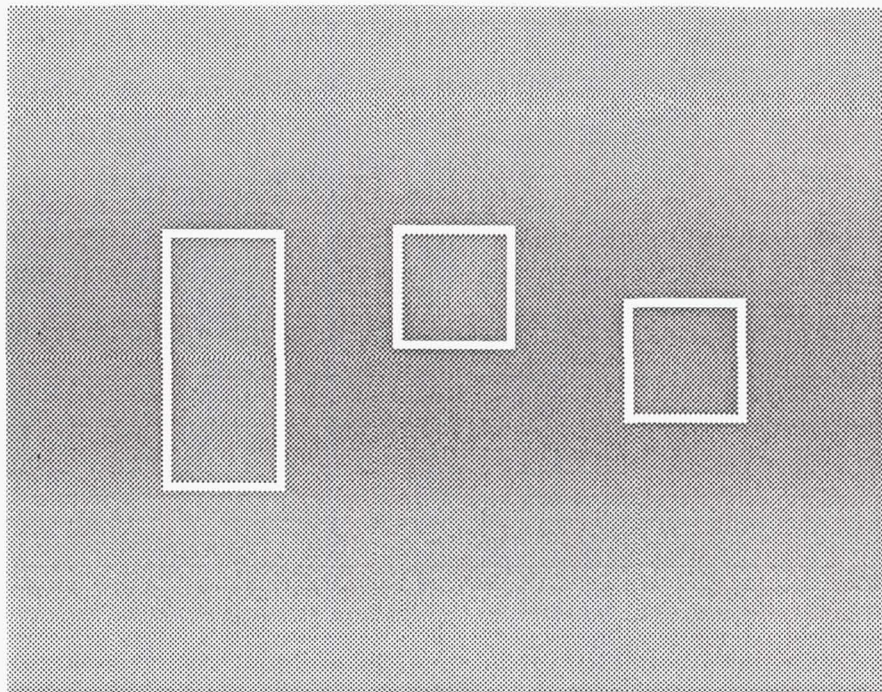
#### ACKNOWLEDGMENTS

The author would like to thank Boeing, Northwest Airlines and United Airlines for their suggestions and help in obtaining the data presented here.

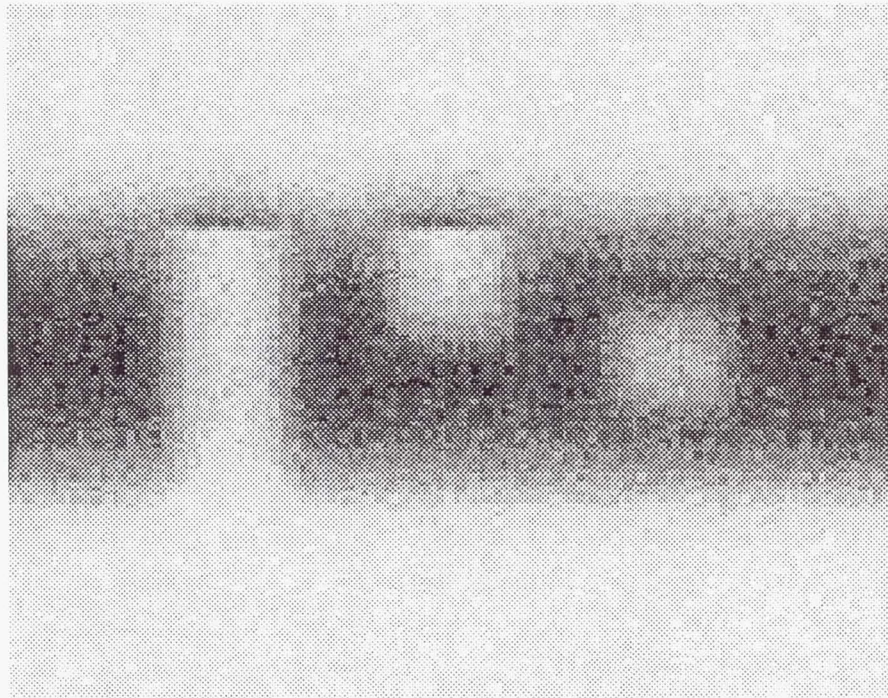
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(a)



(b)

Figure 1. Typical results for simulation of thermal bond inspection system. a. The surface temperature profile for front surface heating and detection. The bond sizes are 2.5 by 7.6 cm and 2.5 by 2.5 cm and highlighted to show positions. b. The time derivative of the surface temperature clearly shows the disbonds.



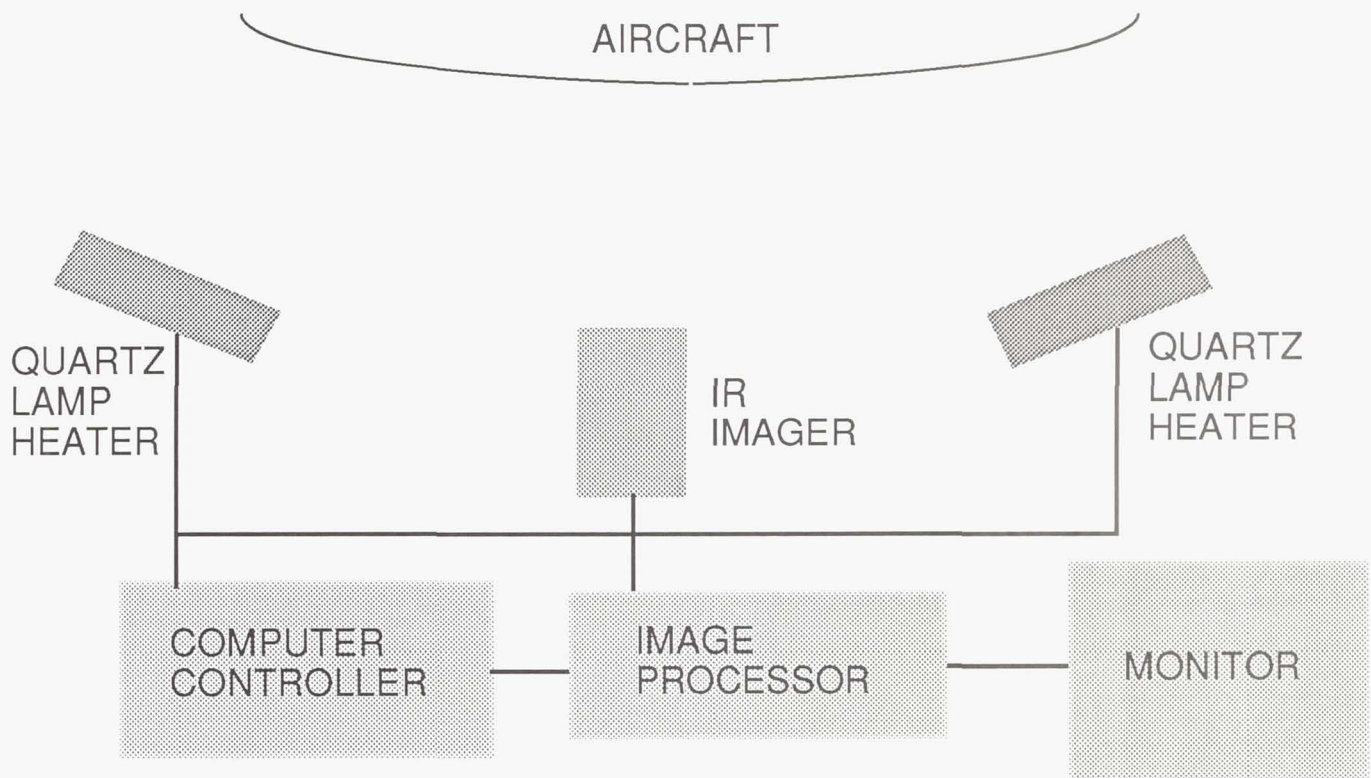


Figure 2. Schematic of thermal bond inspection system develop for field testing of thermographic technique.

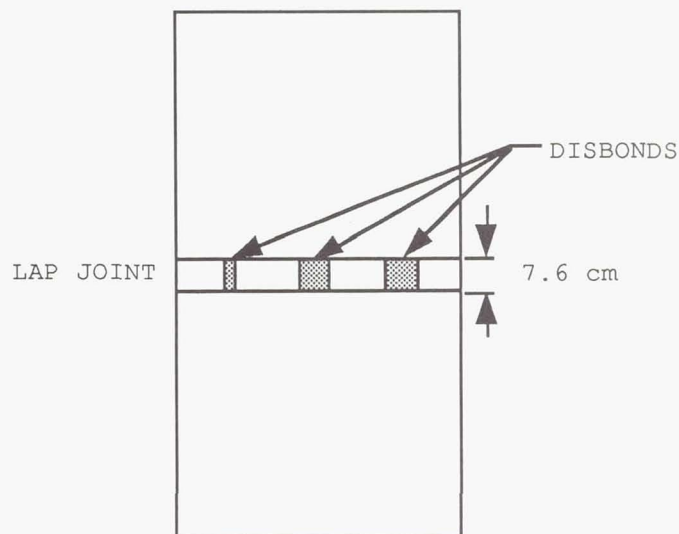
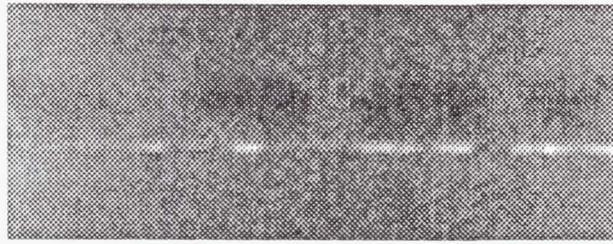


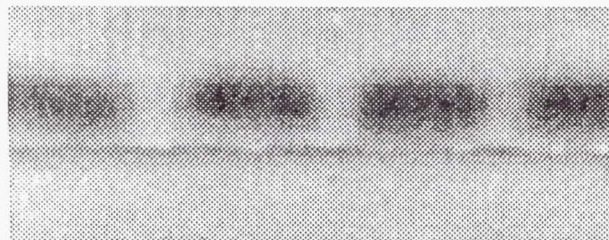
Figure 3. Typical laboratory standard configuration with two aluminum plates bonded with room cure epoxy. Disbonds are created by inserting pull tabs into over lap before bonding. These pull tabs were removed after the epoxy had cured.



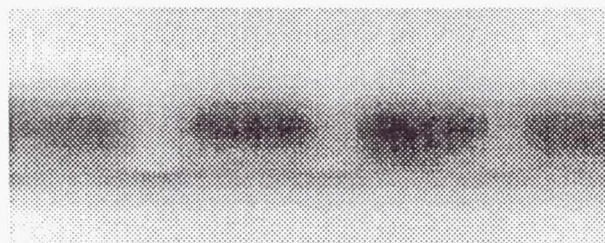
(a)



(b)



(c)



(d)

Figure 4. Typical time derivative images for a laboratory standard with 2.5, 3.8 and 5.1 cm disbonds. Heating was cyclically applied with periods of (a) 5.3, (b) 9.6, (c) 18.1 and (d) 35.2 seconds.



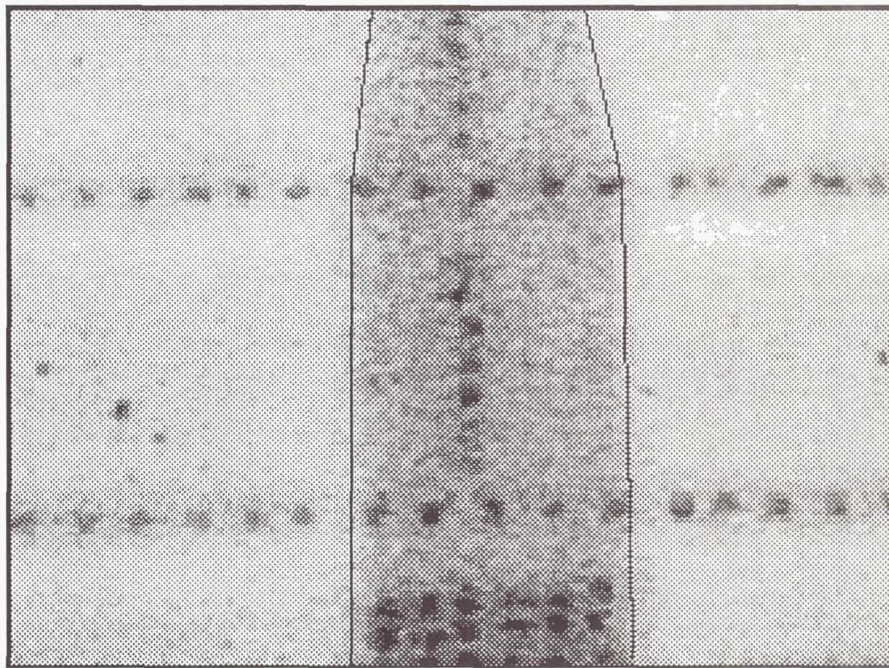


Figure 5. Images of region of airframe with no disbonds. The dark areas show regions of bonding to substructure. The substructure as determined from an engineering schematic of the airframe is outlined on the image.

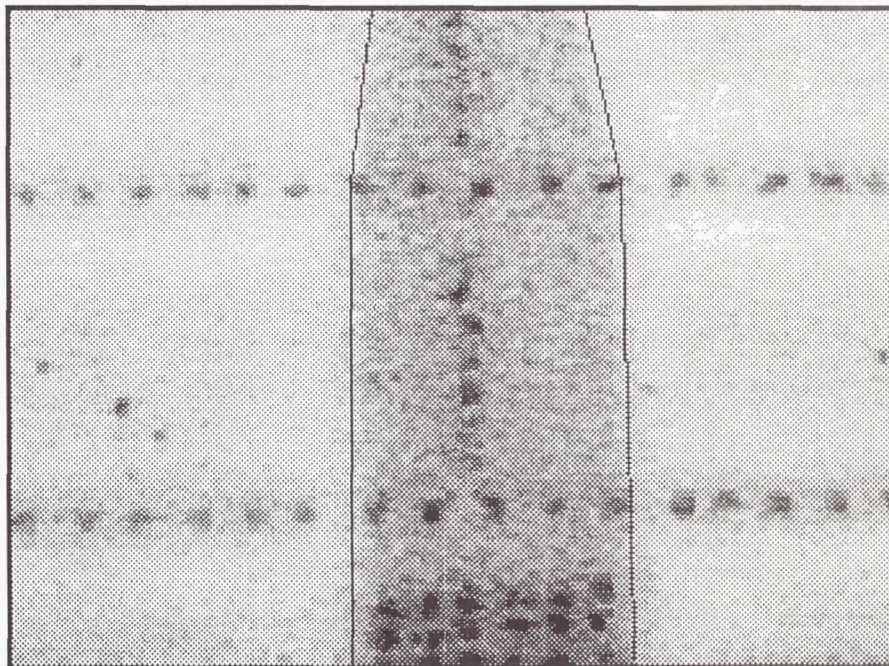


Figure 6. Images of region of airframe with no disbonds. The dark areas show regions of bonding to substructure. The substructure as determined from an engineering schematic of the airframe is outlined on the image.



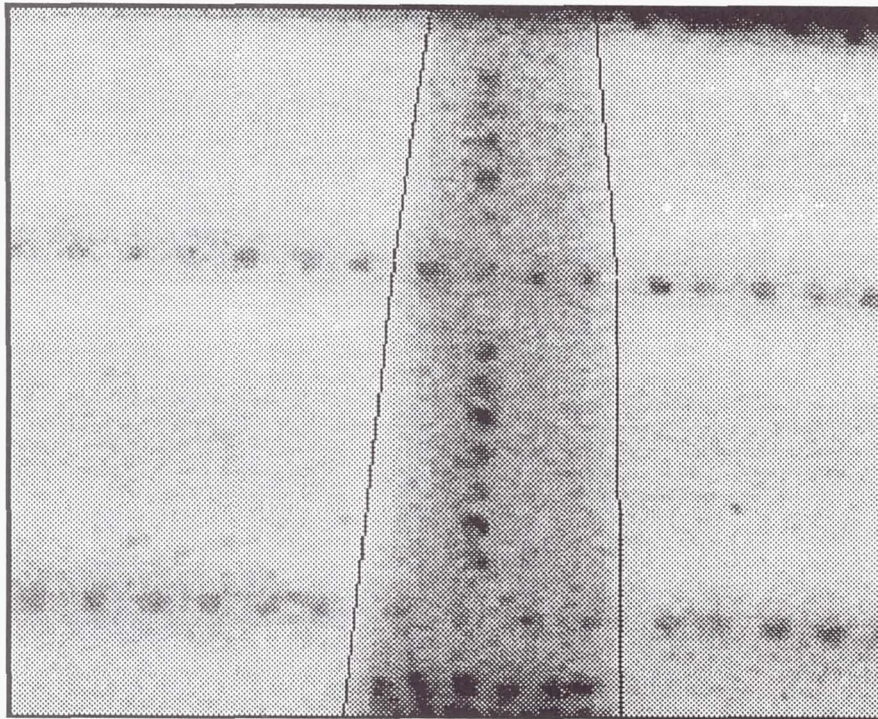


Figure 7. Images of region of airframe with disbonds. The dark areas show regions of bonding to substructure. The substructure as determined from an engineering schematic of the airframe is outlined on the image.

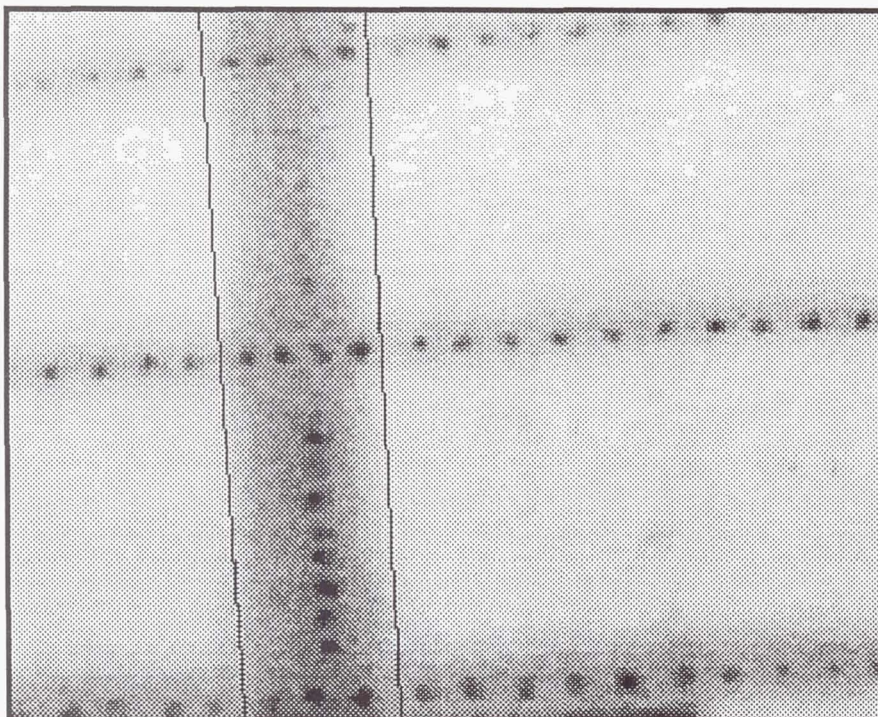


Figure 8. Images of region of airframe with disbonds. The dark areas show regions of bonding to substructure. The substructure as determined from an engineering schematic of the airframe is outlined on the image.



## NDE RESEARCH EFFORTS AT THE FAA - CENTER FOR AVIATION SYSTEMS RELIABILITY

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### SUMMARY

The Center for Aviation Systems Reliability (FAA-CASR), a part of the Institute for Physical Research and Technology at Iowa State University, began operation in the Fall of 1990 with funding from the Federal Aviation Administration. The mission of the FAA-CASR is to develop quantitative nondestructive evaluation (NDE) methods for aircraft structures and materials including prototype instrumentation, software, techniques and procedures and to develop and maintain comprehensive education and training programs in aviation specific inspection procedures and practices. To accomplish this mission, FAA-CASR brings together resources from universities, government and industry to develop a comprehensive approach to problems specific to the aviation industry. The problem areas are targeted by the FAA, aviation manufacturers, the airline industry and other members of the aviation business community. This consortium approach ensures that the focus of the efforts is on relevant problems and also facilitates effective transfer of the results to industry.

### INTRODUCTION

The Center for Aviation Systems Reliability (FAA-CASR) began operation in the Fall of 1990 with funding from the Federal Aviation Administration in response to the Aviation Safety Act of 1988. FAA-CASR was established at Iowa State University because of ISU's long-standing commitment to research, development, application and education of quantitative NDE. FAA-CASR was established as a sister center to the Center for Nondestructive Evaluation which began at ISU in 1984 as one of the National Science Foundation's Industry/University Cooperative Research Centers. The mission of CNDE is to pursue research in nondestructive evaluation in problems of interest to industrial sponsors, to increase the number of students in graduate degree programs with an emphasis on NDE engineering and industrially relevant experience, and to establish a major, national focal point for NDE technology transfer to industry.

FAA-CASR utilizes research staff at ISU as well as subcontracts established with other organizations such as Northwestern University, Wayne State University, and Tuskegee University. The program involves four major projects: 1) Technology Application and Implementation, 2) Engineering Research and Development, 3) Education and Training, and 4) Economics and Management of New Technology. Project 2 utilizes the broad base of NDE knowledge and applies the appropriate technique to various areas of concern in the aviation industry such as fatigue, corrosion, adhesive bonded structures, engine inspection and cracking in airframes and propulsion systems. Project 1 applies the technology and insures its implementation through development of prototype instruments and software, standardization of procedures and techniques, as well as field experiments. In the area of technology transfer, FAA-CASR will work together with Sandia National Laboratory's Aging Aircraft Nondestructive Evaluation Center as part of the FAA research consortium. Project 3 has as its objectives to develop a comprehensive education and training program in the engineering application of nondestructive evaluation and to design a responsive

program that meets the needs specific to the aviation industry through the use of classroom instruction in the theory of the technologies, hands-on laboratory practice, instructional videos and computer simulation of inspections. Project 4 links the human and economic interests into a holistic management approach to development and deployment of new inspection instruments and techniques. A brief description of selected tasks in the program are provided below.

## RESEARCH EFFORTS

### Probability Of Detection Models

The work underway at FAA-CASR is the first approach in which POD concepts have been integrated with actual components within a computer aided design framework to assess field inspection capabilities. The objective of this effort is to demonstrate the use of POD models as an engineering tool to aid in the quantification of the inspectability of aircraft components for eddy current, radiography and ultrasonic techniques. These efforts have their foundation in measurement models that allow the inspection parameters, measurement geometries, and component geometries to be manipulated. This provides unparalleled flexibility and cost savings over the more traditional methods used to acquire POD estimations. In addition, the human component does not enter into the calculations, insuring that decisions are based on instrumentation response allowing the independent assessment of inspector performance and training needs.

The use of POD models offers an inexpensive method to assess reliability of NDE inspection capabilities. They provide a tool for optimizing, verifying and interpreting inspection results from the design stage to in-service tests. Computational models offer a low cost tool for generating test signals for training automated inspection systems and for development of innovative approaches to training as discussed below.

### Eddy Current Field Mapping Instrument

A unique new eddy current probe characterization instrument developed at Iowa State University is being evaluated for calibration and characterization of probes used in aircraft and engine inspections. The instrument uses a laser to map the electromagnetic field of the probe under consideration. This provides an objective, quantitative method for determining the quality and reliability of a probe. Much more information about the probe is obtained than would be possible with calibration on an EDM notch or with measurement of the probe's electrical characteristics. This technique will provide an effective quantitative tool to assess the reliability and sensitivity of eddy current probes used in routine airframe and engine inspections. Presently there is no universally accepted method to do this, resulting in wide variability in probe performance, consequently affecting the reliability and uniformity of eddy current inspections performed.

### Corrosion Detection Techniques

Corrosion continues to be the problem of most concern to the industry. Present detection is based on visual inspection or the use of eddy current or ultrasonics to determine skin thickness. In actuality, the techniques do not detect corrosion or corrosion products but instead determine with limited accuracy the thickness of the outer skin. This does not provide necessary information on second layers. Because of the importance of this problem, FAA-CASR currently has several



techniques under consideration which include x-ray backscatter techniques, x-ray energy sensitive techniques, electrochemical sensors, thermal wave techniques, ultrasonic techniques and eddy current methods.

- X-ray backscatter techniques being explored at Northwestern University offer single sided access and the use of lower energies therefore yielding a faster, less labor intensive technique.
- The use of x-ray energy sensitive techniques is being explored at Iowa State University that allows the characterization and ultimately the quantification of corrosion products which will have a different energy than the "good" material.
- Sensor development underway at Iowa State University is based on a new electrochemical NDE technique to quantitatively detect geometry and composition of corroded areas in lap joints. The technique will offer capabilities to measure metal skin thickness, depth of corroded area, and presence of solid corrosion products.
- Thermal wave imaging techniques under development at Wayne State University offer the possibility for a fast, contactless, single sided access inspection of aircraft structures. This modality is being applied to detection of disbonds, corrosion and cracks around fasteners.
- Efforts are underway at Iowa State University in use of ultrasonic techniques for detection and characterization of corrosion in adhesively bonded structures which includes lap joints, tear straps, honeycomb and composites. Initial effort will utilize squirter techniques with future work extending to contact transducers.
- An eddy current technique developed at Iowa State University for determining the depth and conductivity of layered metal structures is being adapted to determine corrosion-induced thinning of aircraft skin in lap joints. Present eddy current methods for characterizing corrosion in lap joints are only sensitive to thickness of outer layer and are based on empirical calibration of the instrument. The new method is based on swept-frequency measurements of probe impedance over a wide frequency range. This will improve quantitative determination of both skin thickness and air gap between the layers.

#### NDE Detection Of "Hard Alpha" In Titanium Alloys

Titanium alloys are widely used in applications that require high strength at intermediate temperatures, good creep properties and light weight. Much of the demand for these alloys in safety critical areas such as jet engine components. The criticality of these components requires that they be free of defects that could lead to failure. A concern in titanium alloys is the presence of hard alpha inclusions as discussed in FAA Titanium Rotating Components Review Team Report released in December of 1990. Hard alpha inclusions also known as high interstitial defects (HID) are regions of interstitially stabilized alpha of substantially higher hardness than the surrounding material. They are the result of very high localized oxygen or nitrogen concentrations that increase the beta transus and produce a brittle alpha region. These brittle regions then act as stress concentrators in the material and can be a source of crack initiation that eventually leads to failure. Efforts are underway

at the Center for Aviation Systems Reliability and the Center for Nondestructive Evaluation both at Iowa State University to develop a comprehensive set of engineering tools for detection and classification of hard alpha defects in titanium alloys.

### Design And Development Of An NDE Inspection Simulator

Existing NDE measurement models for x-ray, eddy current and ultrasonics will be coupled with computer aided design drawings of relevant aviation components in the development of an NDE inspection simulator. This will allow the user to choose a component, specify the inspection parameters and with the assistance of computer visualization techniques, enable the user to "see" the inspection results. The approach takes the measurement models discussed previously and integrates them in a user friendly computer framework. Field data and reference standards will be incorporated as a complement to the model signals enabling a comparison of model and experimental results. The flexibility of this approach allows part and flaw morphology variation in a cost and time efficient manner. The inspection simulator can be utilized as a training tool for new users and offers a flexible refresher tool for experienced users. The simulator provides a method to insure the inspectability of a component at the design stage and can also be used for optimization of inspection parameters throughout the life of the component. The simulator is being developed in phases with phase I focussing on x-ray.

The inspection simulator is just one of several innovative approaches to education and training being developed at FAA-CASR. A series of NDE videos designed to meet the specific needs of the aviation industry is under way. The cornerstone of the program will be the development of a series of instructional modules that includes both theory and practice components. The approach combines both classroom instruction and hands on practice. The first module proposed for development is a visual inspection module. Guidance in development of this effort will rely heavily on input from industry. This will insure currency and applicability to the true needs of the aviation community.

Participation by FAA-CASR in existing education programs at the University and community college levels is seen as a resource as education and training programs are developed in response to needs of the aviation industry. A one of a kind cooperative educational program is being developed within CNDE in collaboration with Northeast Iowa Community College (NICC) funded by the National Science Foundation. This program is intended to foster and enhance interactions among community colleges, universities and industry as well as to stimulate engineering student interest in NDE. The initial phases of the program include various opportunities for engineering students at ISU and students in the NDT option at NICC to participate in summer intensive NDE related work at CNDE, NICC, and industry. The final objective will be to develop an NDE minor as part of the engineering curriculum at ISU. This program will also serve as a resource for FAA-CASR and the training program that is underway within the FAA-CASR.

### CONCLUSIONS

The Center for Aviation Systems Reliability (FAA-CASR) which is part of the Institute for Physical Research and Technology at Iowa State University began operation in the Fall of 1990 with funding from the Federal Aviation Administration. The mission of the FAA-CASR is to develop quantitative nondestructive evaluation (NDE) methods for aircraft structures and materials including prototype instrumentation, software, techniques and procedures and to develop and maintain comprehensive education and training programs in aviation specific inspection procedures and practices. To accomplish this mission, FAA-CASR brings together resources from universities,



government and industry to develop a comprehensive approach to problems specific to the aviation industry. FAA-CASR utilizes research staff at ISU as well as subcontracts established with other organizations such as Northwestern University, Wayne State University, and Tuskegee University.

The program involves four major projects: 1) Technology Application and Implementation, 2) Engineering Research and Development, 3) Education and Training, and 4) Economics and Management of New Technology. FAA-CASR will also work with Sandia National Laboratory's Aging Aircraft Nondestructive Evaluation Center in transfer of technologies developed to industry. The problem areas are targeted by the FAA, aviation manufacturers, the airline industry and other members of the aviation business community. This consortium approach ensures that the focus of the efforts is on relevant problems and also facilitates effective transfer of the results to industry.

#### ACKNOWLEDGEMENT

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## AGING AIRCRAFT NDI DEVELOPMENT AND DEMONSTRATION CENTER

### (AANC) - AN OVERVIEW\*

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#### SUMMARY

A major center with emphasis on validation of nondestructive inspection techniques for aging aircraft, the Aging Aircraft NDI Development and Demonstration Center (AANC), has been funded by the FAA at Sandia National Laboratories. The Center has been assigned specific tasks in developing techniques for the nondestructive inspection of static engine parts, assessing inspection reliability (POD experiments), developing test beds for nondestructive inspection validation, maintaining a FAA library of characterized aircraft structural test specimens, and leasing a hangar to house a high flight cycle transport aircraft for use as a full scale test bed.

#### INTRODUCTION

The three principal government sponsored locations where research on application of nondestructive inspection (NDI) to aging aircraft is occurring are at the National Aeronautics and Space Administration (NASA), Langley, VA, the Center for Aviation Systems Reliability (CASR) at Iowa State University, and at the Aging Aircraft NDI Development and Demonstration Center (AANC) at Sandia National Laboratories (SNL), Albuquerque, NM. The last two centers are under the sponsorship of the Federal Aviation Administration (FAA). The AANC was established August 6, 1991. Initial funding for this Center was \$3.4M with its projected scope a minimum of \$15M over 5 years. Sandia is the prime contractor and has established contractual relations with Science Applications International Corporation (SAIC) and New Mexico State University (NMSU) who are partnering in this Center. Initially, SAIC is involved with the NDI of static engine parts as well as probability of detection (POD) studies, and NMSU is involved with coherent optic inspection techniques. Both of these activities are discussed later.

The fact that these centers have been established is, at least in part, at the request of FAR 121 operators, repair modification centers, transport and general aviation manufacturers (foreign and domestic), and other government agencies such as NASA and NTSB.

\*This work was supported by the FAA Technical Center, Atlantic City International Airport, New Jersey.

Recommendations (ref. 1) from the 1988 International Conference on Aging Airplanes included: "the FAA should sponsor research and development for improvement of NDI technology." Additional specifics associated with this recommendation were: "the FAA would acquire a test bed and using that vehicle would evaluate existing and new NDI methods with full participation of the operators and manufacturers." This conference, which was held shortly after the Aloha accident, was not the first time that a recommendation had been made that additional NDI resources be applied to aircraft inspection. After a crash of a cargo flight enroute to Lusaka, Zambia, on May 14, 1977, Flight International (ref. 2) quoted: "Big advances have been made in non-destructive testing (NDT) techniques. High frequency and low frequency eddy current probes, x-rays, ultrasonics, acoustics, dye-penetrants, and so on are a long way from the white coated inspector with his torch and magnifying glass. He is still the most valuable detective, but he needs help as structures get more complicated. He has nothing to cover the whole aeroplane, including its innermost recesses, for incipient cracks." The cause of the crash in Zambia was determined (ref. 3) to be due to the fact that the aircraft's right hand stabilizer rear top chord spar had failed prior to flight due to fatigue.

It is the intent of the AANC to address many of the NDI related issues of the preceding paragraph. Mindful of the recommendations contained in reference 1, full participation of the manufacturers and operators is both welcome and encouraged. This paper will clearly define the goals and current technical activities of the AANC.

#### GOALS OF THE CENTER

It is first appropriate to discuss the resources that SNL brings to the AANC which hopefully will in time benefit the manufacturers and operators. Sandia is an engineering laboratory. Its prime mission is to design, develop, and test nuclear weapon systems. These systems are placed into production in adherence to a strict time schedule. Subsequent quality in the weapon stockpile is monitored over the life of the weapon program (which is comparable to the lifetime of an aircraft). Three thousand engineers and scientists have advanced degrees in such requisite disciplines for aircraft inspection as NDI, reliability and statistics, human factors, structural and fracture mechanics, corrosion, robotics, and image processing. More important, SNL is a national laboratory, and therefore, has the potential to serve as a neutral party between the regulators, manufacturers, and operators. The principal goal of the AANC is to satisfy its customer, the FAA, while gaining acceptance from the manufacturers and operators as a contributor to their respective industries. These contributions must recognize both the need for reliability in their application to aircraft inspection and the competitive nature of these industries (i.e. they must be cost effective!).



Specifically, SNL's agreement with the FAA calls for three phases associated with its program. Phase 1 (complete in six months) includes initiating tasks defined by the FAA, recommending pilot NDI projects to the FAA, and providing test bed definition. The tasks defined by the FAA include both developing and implementing methodologies for POD studies for select inspection techniques for transport and commuter aircraft. Also included in the FAA defined tasks is the establishment of a basis for the evaluation of static engine parts. It is interesting to note that these tasks, along with the definition of test beds for NDI technique validation, all resulted from recommendations from the 1988 International Conference on Aging Aircraft. It is envisioned that pilot projects proposed by the AANC will be submitted to a manufacturer/operator screening committee before submittal to the FAA for approval. Phase 2 (complete in 18 months) involves the initiation of technology transfer to the operators based on results from phase 1, as well as continuation of phase 1 projects including test bed implementation. Phase 3 (complete in 5 years) involves continuing technology transfer, pilot automation inspection projects, and implementation of multiple inspection modalities to increase flaw POD.

An FAA library of characterized test specimen, now at the John Volpe Transportation Systems Center, will be housed at the AANC and used for NDI technique validation. The number of test specimen will be increased as required to fulfill industry requirements. Test specimen will be made available to industry on a limited basis to be determined by the AANC or at the request of the FAA. These specimen, as well as the test beds, will be housed in a hangar to be leased as soon as possible at the Albuquerque International Airport.

#### CURRENT CENTER ACTIVITIES

The AANC is functionally organized with staff in place in the structural, inspection reliability, and NDI (radiography, coherent optics, ultrasonics, visual, thermography,...) areas. Sandia consultants in fracture mechanics, corrosion, and robotics have been identified. A Navy lieutenant has been assigned to the program for a three-year period to help identify and transfer cost effective technology from the DoD to the AANC. Contracts are in place with both SAIC and NMSU for their support. Another contract is currently being negotiated with the City of Albuquerque for a hangar. That hangar contains approximately 17,000 square feet of floor space and 10,000 additional square feet of office/storage space. It is located next to a large aircraft maintenance facility.

The hangar is envisioned as a location to validate various NDI techniques for their applicability to the FAA's aging aircraft initiative. The hangar will house various test beds, the test specimen library, and hopefully a high flight cycle aircraft of the McDonnell Douglas DC9 or Boeing 737 size category. Aircraft usage requirements are currently being defined. The test specimen library has been requested from the John Volpe Transportation Systems Center and should soon be shipped to SNL.

Initial focus of the AANC has been on configuring itself to be a long term resource for the FAA. While the average AANC member has over 20 years experience, each is acutely aware of the need to familiarize himself with the FAA regulatory system, airframe structures, and inspection facilities. Outside consultants have been hired to provide training in damage tolerance analysis and corrosion in aerospace structures. The FAA has been consulted through its Technical Center, National Resource Specialists, Principal Maintenance Inspectors (PMIs), and local Flight Standards District Offices (FSDOs). AANC members have visited both Boeing and McDonnell Douglas as well as McClellan AFB. Several trips have been made to Mesa Airlines where members were extended the courtesy of spending one day observing tear downs of Pratt and Whitney PT6 engines and one entire night observing structural inspections in their maintenance hangar. U S Air has been briefed on AANC activities and has invited Center members to participate in D checks of their aircraft. AANC members spent one day with flashlights and a FAA corrosion expert inspecting a 1963 vintage Boeing 707 at the USAF Phillips Laboratory to gain needed experience. Numerous manufacturers of NDI inspection equipment have contacted the AANC. All of these contacts are welcome and it is recognized that more are required.

A significant amount of commercial NDI hardware is available in industry which is not being implemented in maintenance programs by the operators. This is due both to lack of proven effectiveness of this hardware through an extensive evaluation program and to lack of approval by the FAA for inclusion in operator maintenance programs. One objective of the AANC is to provide hardware evaluation to complement that of the manufacturers and operators. Where evaluation results are encouraging, and additional improvement is required, technology transfer and limited funding to equipment manufacturers can be provided. The objective is not to "reinvent the wheel" but to take advantage of existing technology. Some examples which illustrate potential technology which could be considered include:

Advanced Shearography - A laser based optical technique for full field video imaging of subsurface flaws identified through changes in strain (Laser Technology, Inc.).

Autoscan Flaw Detection - An ultrasonic scanner using shear waves and operating in the pulse-echo mode for fatigue cracks of rivets (Systems Research Laboratories, Inc.).

DiffRACTO-Sight or D-Sight - An optical technique used for visualizing surface distortions created by a change in surface topography greater than 10 micrometers (DiffRACTO Ltd.).

Fiber Optic Strain Sensors (FIBERTRONIC System) - A fiber optic strain sensor which uses optical time domain reflectometry to identify structural deflections and locations due to cracks and delaminations (G2 Systems Corporation).



Flight Loads Recorder - A microprocessor based recording system capable of recording flight structural parameters (ESPRIT Technology, Inc.).

Magneto-Optic/Eddy Current Imager (MOI) - An eddy current inducing and magneto-optic sensing element for detecting cracks and corrosion through real-time imaging (PRI Instrumentation).

Mobile Automated Ultrasonics Scanner (MAUS) - An ultrasonic and eddy current scanner providing a fast pseudocolor image of the scanned surface (McDonnell Douglas).

In addition to these technologies, one can readily conceptualize other productive areas for contributions. Inspectors flashlights could be improved to enhance visual inspections. Eddy current probes could include hybridized electronics and transmit their signals via a two-way rf data link to a personal computer for analysis and storage. This probe design would enhance human factors by eliminating bulky impedance scopes and/or meters, enhance data analysis, and be cost effective to implement. Additional examples exist.

An initial project assigned to the AANC involves an investigation of the POD associated with various inspection techniques. POD curves are an integral part of damage tolerance analysis in that they are necessary for deriving rational inspection intervals. The valid estimation of the POD requires that the data be taken under the conditions (including people, facilities, equipment, and procedures) in which aircraft are actually inspected. For eddy current lap splice inspections, the POD is currently estimated using 14 year old data taken across Air Force facilities or from laboratory experiments.

The AANC has initiated a program to design an experiment that is capable of providing the required data for estimating POD curves applicable to eddy current inspection of lap splices. The primary concern of the program is to identify and address "human factors" issues. Human factors have to be considered not only as part of the target study (i.e. their influence on the inspection results), but also in assessing to what degree the experimental conditions can be taken as an adequate model of the inspection process. From this initial POD study, it is hoped to expand to other inspection techniques (e.g. visual). This work is intended to support both transport and commuter type aircraft.

Another task assigned to the AANC by the FAA involves the NDI of static engine parts. This FAA assigned task, mentioned earlier, was derived from inputs at the first International Conference on Aging Airplanes. This conference stated the need for the development of enhanced inspection needs for static engine parts (cases and static structure).



An actuarial analysis of the JT8D engine was undertaken by SAIC to identify static engine components which caused engine shutdowns and unscheduled engine removals using FAA Service Difficulty Reports and Air Carrier Aircraft Utilization and Propulsion Reliability Reports. The JT8D component failure pattern over a two-year period identified three components for which enhanced NDI procedures may be beneficial. One of the failures was cracking in the weld of the fuel drain bosses of the outer combustor case. The other two failures were cracking in the #6 oil bearing tube and in the 13th stage bleed air duct. An enhanced NDI ultrasonic scanning technique was developed for detecting the cracks in the weld of the fuel drain boss.

As an extension of this work, SNL contracted SAIC to perform three tasks. Task one directs SAIC to develop NDI procedures for detecting cracks in the #6 oil bearing tube and the 13th stage bleed air duct of the JT8D engine. Task two instructs SAIC to investigate whether cracking of the weld of the fuel drain bosses of the outer combustor case of the JT9D and CF6 engines is a problem. If a problem also exists in the latter two engines, SAIC will develop an ultrasonic scanner to inspect these welds. A field demonstration with the engine on wing is planned for the ultrasonic scanning technique. The third task is to carry out the actuarial analysis retroactively for a 36 month period for the JT9D, CF6, and PT6 engines to identify any static engine components for which advanced NDI procedures may be beneficial. The work is to be completed in one year.

#### CONCLUSION

Work assigned to the AANC by the FAA in NDI reliability (POD studies), inspection of static engine parts, and test bed development is in process. The test specimen library has been requested from the FAA for shipment to the AANC. A hangar to house the AANC is in the process of being leased. Coordination with airframe and engine manufacturers, airline operators, NDI equipment manufacturers, and the DoD is occurring. Technical suggestions for and interactions with this Center by manufacturers and operators is welcome and solicited.

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## NONDESTRUCTIVE INSPECTION PERSPECTIVES

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### SUMMARY

This paper presents ideas for consideration by those concerned with commercial aircraft nondestructive inspection (NDI). The perspective is that of an individual with a background in military aircraft NDI, and important differences are indicated between the commercial NDI and military NDI activities. In particular, it is significantly more expensive to implement some new NDI technology, and therefore, in-depth cost-benefit studies for commercial users are recommended.

### INTRODUCTION

Nondestructive tests have greatly improved in their ability to reliably reveal hazardous defects in aircraft structures. In addition, new testing modalities have become available to our arsenal of NDI methods. However, no test is clearly so self-sufficient that it can be relied upon by itself, even for a single type of defect. Sensitivities (probability of detection for critically significant defects) and specificities (probability that a diagnosis is correct) are functions of equipment, procedure and personnel reliability which cannot be assumed to approach 100%. Moreover, there is a great diversity of defect types and geometric configurations. Therefore, use of a single method for a single type of defect (such as stress corrosion cracking or exfoliation corrosion) is only of moderate value. Conversely, for principle structural elements and other important components of those aircraft at highest risk, the use of a full spectrum of complementary nondestructive inspection methods should be of high value. The objective in this paper is to address improving NDI technology for commercial aircraft, and to ask if the military NDI full spectrum approach contains lessons worthy of in-depth consideration by the commercial aircraft sector.

### NEW NDI TECHNOLOGY RESULTS

Sacramento Air Logistics Center (SM-ALC) is one of five U. S. Air Force maintenance centers. The U.S. Navy also operates its own aircraft maintenance depots. At SM-ALC, the responsibilities of the NDI division include the inspections of:

1. intact aircraft brought in for spot checks of critical items (Analytical Condition Inspection);

2. intact aircraft and/or detached components of aircraft brought in for major maintenance, typically at four or five year intervals (Programmed Depot Maintenance);
3. components used to supply replacement parts for depot or field maintenance (MISTR).

In the past decade, at least sixteen major advances in NDI technology have been carefully evaluated for possible implementation at SM-ALC (Table 1). The studies involve complex subjects such as advanced computer engineering, laser physics and nuclear science. The goal was to provide cost-benefit comparisons of the options available by teams independent of equipment vendors. Technologies evaluated and found not to be required for implementation at SM-ALC at this time, include: laser holography, shearography, infrared thermography, x-ray backscatter, x-ray computed tomography, dual energy x-ray, neutron backscatter, and a transportable neutron radiography system tested in cooperation with the U.S. Navy. The fact that no follow-up action is planned at SM-ALC on these technologies at this time should not be interpreted to mean there will be no further interest.

The technology upgrades of most significance that have been implemented are as follows:

Advanced Eddy Current Technologies. Eddy current is recognized at SM-ALC as an extremely powerful and versatile technology and a continuous effort is made to utilize the most advanced technology. Computer signal processing is playing an increasingly important role in this field. The range of eddy current technologies implemented includes impedance type testers for surface and subsurface crack detection, phase/amplitude type testers for corrosion cracks, thickness indication, and fastener and bolt hole inspections. In addition to our extensive microprocessor controlled eddy current equipment inventory, we have auxiliary probes and transducers valued at over \$150,000. We use commercially available equipment, we manufacture some equipment to meet our specific needs, and we keep abreast of computer software developments in eddy current technology taking place elsewhere, such as the current Air Force sponsored work being pioneered by Northrop.

Advanced Ultrasonic Techniques. In 1982, we introduced into service a twelve axis robotically controlled water squirter ultrasonic scan system which is used primarily in through transmission mode for locating disbonds in detached components. The unit interfaces with five different computers.

Microfocus Real-Time X-ray. Because microfocus x-ray technology provides a geometric magnification of the image on the detecting screen the resolution limitations of electronic imaging become acceptable. Two such real-time x-ray systems were put into service at McClellan Air Force Base about 1985 and computer-based image



analysis has produced standard routines which minimize variations between individual radiographers.

Reverse Geometry X-ray. Instead of the normal point source and planar detector, this technology uses a planar source which is swept by the electron beam near the object, and a single point digital detector at a distance on the far side of the object. Because the point detector does not receive scattered x-rays the system offers exceptional signal to noise ratios. Images displayed on a video screen can be given a three dimensional effect by use of a second detector and the use of polarized glasses.

Film Digitization System. Equipment is being implemented that will digitize the data from standard 14" x 17" x-ray film, permit electronic image processing, and provide archival storage on optical discs. This is a step towards reduction in film archiving problems and towards an eventual Inspection Data Interpretation System (IDIS) that will provide a network for a comparative interpretation of formatted images from all complementary NDI systems.

Intact Aircraft X-ray. This system consists of a shielded bay in which aircraft up to about 78 feet wing span can be scanned radiographically using a programmable overhead gantry positioner. A yoke system holds the x-ray source - either 160 Kv microfocus or standard focus 320 Kv - on one side of the aircraft panel, and the real-time x-ray imaging device on the opposite side.

Intact Aircraft Neutron Radiography. The neutron inspection bay is adjacent to the x-ray bay, and is similar in size and mode of operation. However, in this case, the radiation source is a californium-252 isotopic source of neutrons. There are, in fact, two similar sources, one used with the overhead gantry robot for electronic imaging of flight control panels and the other used for a ground based positioner to perform film radiography of underside fuselage panels.

One advantage of the intact aircraft x-ray and n-ray systems is that they can inspect aircraft with minimal out-of-service time (i.e., spot checks or Analytical Condition Inspections on aircraft not required to undergo major overhaul). A typical scan can be completed in a total of four days, two days in the x-ray and two days in n-ray.

The systems entered operation in 1990 and performance equals or exceeds the design criteria. The first ten aircraft showed over 240 incidents of hidden moisture or honeycomb corrosion; critical information that was not indicated by other means.

Detached Component High Resolution Neutron Radiography. The Stationary Neutron Radiography System (SNRS), which has a small nuclear reactor as a neutron source, is designed to perform high



resolution electronic or film imaging at a high throughput rate on detached components. There are four bays which can be used simultaneously. Three of them are equipped with robotic positioners and real-time neutron imaging equipment for parts up to a 37 foot long wing, and the fourth bay is available for inspection of pyrotechnic devices, hydraulics or turbine blades.

This system entered operation in mid-1991 and performance equals or exceeds expectations.

## DISCUSSION

A question is whether experience being gained in the application of new NDI technology for aging military aircraft deserves in-depth evaluation for possible application to commercial aircraft?

Certainly the history of NDI technology development teaches us to be looking for new advances. At SM-ALC in 1940, we had only x-ray and magnetic particles. Eddy current was introduced in 1963, and the first ultrasonics was in 1967. The technology changes currently underway will be neither the first nor the last.

The motivations for improved NDI at SM-ALC include: (1) extension of aircraft life, (2) reduction of risk of in-flight mishap, and (3) an improved database for engineering activities. These are motivations that may be also of interest to the commercial aircraft sector, but specific cost benefit analyses would be necessary before conclusions could be reached.

TABLE 1	
NEW METHODS EVALUATED 1980 - 1990	
WITHOUT FOLLOW-UP PLANS	SELECTED FOR IMPLEMENTATION
Holography	Eddy Current - Computer Advances
Shearography	Ultrasonics - Computer Advances
Thermography	X-ray Microfocus - Real-Time
X-ray Backscatter	X-ray - Reverse Geometry
X-ray Computed Tomography	X-ray - Film Digitization
X-ray Dual Energy	X-ray - Intact Aircraft
X-ray Backscatter	N-ray - Intact Aircraft
N-ray Transportable	N-ray - High Resolution - RT

Current and Future Developments in Civil Aircraft  
Non-destructive Evaluation from An Operator's Point of View

Jeff Register  
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### INTRODUCTION

In June 1988, the first International Conference on aging aircraft was held to address NDT aging aircraft issues. From this meeting, a research program was initiated and funded by the FAA. As a result of this program, a lot of work has been done to study current NDT practices in the aviation industry and secondly, to research and develop new NDT methods to improve the reliability and efficiency of in-service inspection of aircraft structures and powerplants.

The following is an overview of current and future developments in civil aircraft NDT, as viewed by an air carrier and the concerns for NDT in the future.

The current philosophy at Northwest Airlines is best represented by Aircraft Maintenance Operation's Vision and Mission Statement as shown below:

### OUR VISION

Everywhere in the operation everyone will see and feel a difference. Our facility and work place will be **safe**, clean, and well-organized. There will be a noticeable sense of **pride** and **trust** where all are challenged professionally and enjoy coming to work. A strong spirit of working together will be evident, embraced with constant unrestricted **communication** and **performance development**. A **commitment** to empowering individuals will be obvious as work groups concentrate on their customers and measure their own performance.

The organization will be lean and effective, conveying a real commitment to **trust**, **respect** and **training**. Everyone will be recognized for their **leadership** and **vision**, with a focus on continuous **improvement** and **innovation**. The leaders will be honest, competent, and forward-looking, while being inspiring teachers, coaches and change-makers.

There will be an obvious interest in people, their needs and well-being, their community involvement and development-through support, encouragement and **recognition**. We will have a labor-management committee for early **problem solving** and **creative solutions** within the bounds of the contract.



The Aircraft Maintenance Operations group will have a reputation for producing a quality product expeditiously and efficiently, while controlling costs and reducing waste. This group's accomplishments will be the example within the company and the industry that others will strive to attain.

This dynamic, forward-thinking vision will continue to challenge our strategic direction, stretching the organization to new levels of excellence.

#### **OUR MISSION**

To provide safe, clean, reliable aircraft from major maintenance to support Northwest Airlines' strategic operating plan.

We at Northwest Airlines feel it is imperative to have a Vision/Mission Statement in order to institute a dynamic structure to drive Operational Process Improvement and to develop an interactive communication process. This in turn will help to identify, analyze, and pursue opportunities to improve work processes.

As a part of the research programs, the airlines have been host to many groups of people conducting research, in order to assess the conditions of the "actual" test environment or the "real world". It soon became apparent that many of these groups did not understand what the NDT process consisted of and what events drove the airlines into performing a NDT Inspection.

The following is an overview of the four events which initiate NDT Inspection on aircraft:

- (1) Service bulletins (S/B) / Airworthiness Directives (AD's)
  - Service Bulletins
    - S/B Original Equipment Manufacturer (OEM) generated inspections in which compliance is not mandatory.
  - Airworthiness Directives
    - AD FAA mandated/approved inspections (usually generated from S/B).
- (2) Damage Assessment / Repair Assessment
  - Assessment of damage on aircraft prior to repair/rework or replacement.
  - Monitor temporary repairs until permanent repair is accomplished.
  - Repair assessment program (RAP). Assessing existing repairs on aircraft.
- (3) Monitor Inspections
  - Monitor existing defects on aircraft.
  - Monitor defects existing in repair area.



(4) Operator Generator Inspections

- Confirmation of indications found by other methods of inspection (visual).
- Additional inspections conducted by operator which are not required by OEM or FAA.

Of the four events listed above, items (1) through (3) are all performed in accordance with OEM's NDT Procedures of Inspection methods generated by operators and approved by OEM. Item (4) in most cases also uses the above NDT criteria.

### OVERVIEW OF CURRENT NDT METHODS

Current methods used by operators include:

- Magnetic Particle
- Dye Penetrant
- Radiography (X-ray/Isotope)
- Ultrasonic
- Eddy Current

Equipment associated with these methods are generally portable for ease of use in a hangar or flight line environment.

Method of choice depends on a multiple of factors including: application, access, material type, speed, environment in which the test is to be performed, power requirements, etc. Of the three most popular methods of choice (radiography, ultrasonic and eddy current), eddy current is usually the most preferred method due to its simplicity of use, ease of interpretation and speed. Ultrasonics would follow eddy current due to its sensitivity in detecting small flaws. This sensitivity, at the same time, causes difficulty in interpretation. Next would be radiography. Radiography reduces the amount of opening time required for visual and other methods of inspection, but is very expensive due to the cost of the equipment and hangar down time.

## FUTURE CONCERNS/DEVELOPMENTS FOR NDT

The following is a list of concerns from an operator's point of view, which need to be addressed as a part of the aging aircraft research program.

Research needs to be a coordinated effort between the operators, research facilities, NDT equipment manufacturers, OEM's, and other industries (i.e. military, nuclear research) in order to produce reliable NDT techniques and valid studies.

Composite inspection techniques need to be researched and developed in order to address future inspection demands for advance composite components. Major structural composite items will require NDT inspection to detect disbonds, moisture intrusion and impact damage.

Training programs need to be standardized to establish consistency throughout the aviation industry. Programs should consist of formal classroom training, which is tailored to the aviation industry. Formal training should also include other conditions which effect the results of an inspection, such as: temperature of item tested, damaged area, areas previously reworked, different configurations found on aircraft. The inspector must have the knowledge and the capability to adapt to these conditions in order to perform a valid test.

## CLOSING REMARKS

Implementation of advanced NDT methods developed through programs like these and their success will depend largely on a cooperative effort between OEM's - equipment vendors - research facilities - and aircraft operators. New NDT methods must then be validated on actual aircraft, approved by the OEM and implemented into its inspection program. This process must be completed before the operators can utilize such methods to comply with AD's required on aircraft.

Cooperative efforts such as these will assist operators to continue to provide safe aircraft to the flying public.



Ageing Airplane  
Repair Assessment Program  
for Airbus A300

J.M. GAILLARDON, H.-J. SCHMIDT and B. BRANDECKER

**SUMMARY** As a result of a commitment between the FAA and ATA the Airworthiness Assurance Task Force (AATF) steering committee requested the transport aircraft manufacturers to develop a program to assess the repairs on in-service aircraft since recent Supplemental Structural Inspection Programs (SSIP) had not been addressed to assessing repairs. The intent of the assessment program is to remove all existing ambiguities and to achieve a sufficient level of safety across the fleet by applying actual fatigue and damage tolerance criteria defined by FAR 25.571 to repaired structures. A general three stage program has been developed with major aircraft manufacturers and airlines cooperation including area/component classification, repair categorization and inspection/removal requirement establishment which is the basis for model specific repair assessment programs.

This paper describes the current status of the repair categorization activities and includes all details about the methodologies developed for determination of the inspection program for the skin on pressurized fuselages. For inspection threshold determination two methods are defined based on fatigue life approach, i.e. a simplified and a detailed method. The detailed method considers 15 different parameters to assess the influences of material, geometry, size, location, aircraft usage and workmanship on the fatigue life of the repair and the original structure. For definition of the inspection intervals a general method is developed which applies to all concerned repairs. For this the initial flaw concept is used by considering 6 parameters and the detectable flaw sizes depending on proposed non destructive inspection methods. An alternative method is provided for small repairs allowing visual inspection with shorter intervals.

## 1. INTRODUCTION

As a direct result of several recent incidents on civil transport aircraft the FAA has launched several programs to assess aging transport aircraft, among others the assessment of existing repairs.

The Airbus repair assessment program is shortly described and comprehensive information is provided about the methodologies for assessment of repairs at Airbus A300 aircraft.

The main objective of these methodologies is the assessment of existing repairs on in-service aircraft, i.e. the determination of necessary inspection thresholds, and intervals and/or time of modification or replacement for the repair. The assessment of each repair includes static as well as fatigue and damage tolerance aspects.

## 2. REPAIR ASSESSMENT PROGRAM

### 2.1 Reason for Repair Assessment Program

Besides the recent incidents leading to a general concern, there are other technical and regulatory aspects for launching the assessment of repairs. Most of the repairs in service were designed for static and fail-safe criteria for aircraft designed prior to amendment 45 of FAR25. The SSIP programs brought the aircraft designed pre-amendment 45 of FAR 25 to a status according to amendment 45, but the repairs were not addressed. Due to this aspect and under consideration of the age of the fleet the damage tolerance behaviour of the repairs has to be analyzed. In addition some repairs hide the primary structure to such an extent, that specific inspections may be required. Furthermore the interpretation of the earlier repair instructions

may be difficult or the repair had not been performed in accordance with the instructions due to several reasons. At last the design principles and the justification methods of repairs have been evolved from entry into service of Airbus A300 till now, i.e. within approximately 20 years.

### 2.2 Objectives of Repair Assessment Program

The objectives of repair assessment program for the Airbus A300 fuselage, are clearly defined:

1. Demonstration of damage tolerance capability of repairs and surrounding structure
2. Assessment of repairs by the operators without complex analysis using guidance material supplied by Airbus Industrie
3. Establishment of appropriate actions for each repair, i.e. definition of inspection requirements and/or removal/modification limits.

### 2.3 Repair Categorization

The AATF task units have developed a general flow diagram, which shows the operator how to categorize the repairs and to apply the appropriate actions. Figure 1 shows a three-stage program including the application of the manufacturer's guidelines. Stage 1 assessment is followed by Stage 2 for principle structural element (PSE) areas which will be defined in the Structural Repair Manual (SRM). For non-PSE areas the normal maintenance program is considered to be sufficient. The Stage 2 assessment leads to one of the four categories with the following definitions:



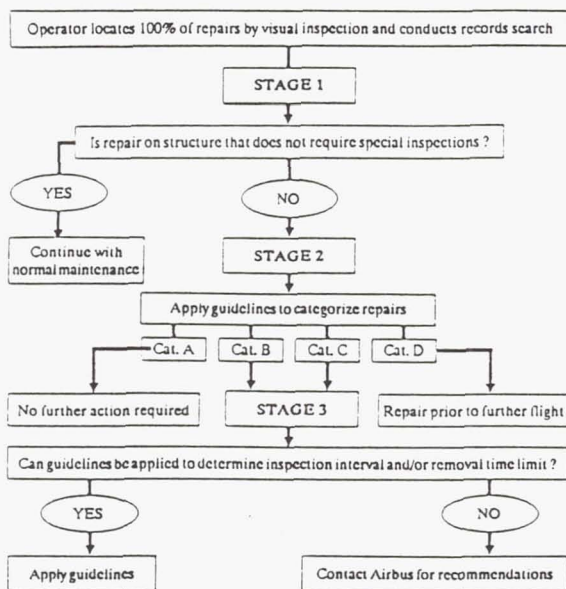


Figure 1: Repair assessment program

#### Category A

Meets the intent of the design certification basis of the airplane. Requires no special inspections other than normal maintenance.

#### Category B

Meets design certification basis of the airplane; however must be periodically inspected beyond normal maintenance requirements to ensure structural integrity.

#### Category C

Meets design certification requirements of the airplane; however, repair is clearly of a temporary nature. Structural integrity requires periodic inspections other than normal maintenance and repair must be replaced or upgraded to a category B or better at a certain time limit.

#### Category D

Does not meet design requirements and/or exhibits structural degradation. Must be upgraded to a category C or better by replacement or repair, before further flight.

The category A is achieved for repairs with sufficient static strength located in low stressed PSE areas which will be defined by Airbus Industrie. Category D applies mainly to repairs with marginal static strength or not designed according to the state of the art or being in a bad condition.

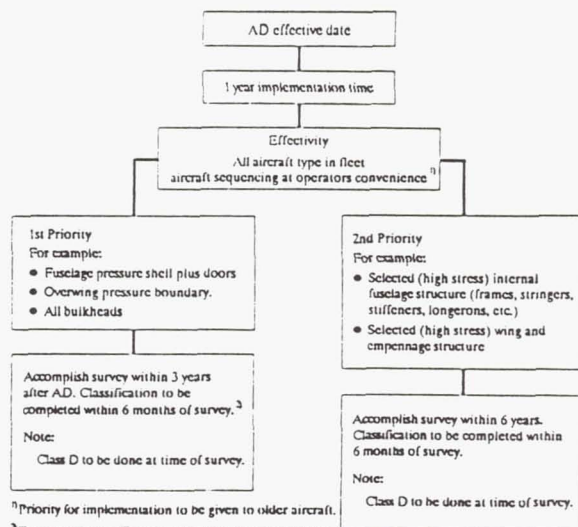
The Stage 3 assessment is applied for repairs categorized to B and C, i.e. establishment of inspection requirements and/or definition of replacement/modification time.

The manufacturer's guidelines for categorization in Stage 2 and the Stage 3 assessment which have been developed for the Airbus A300 are described in detail in the next chapters.

#### 2.4 Evaluation Time Frame

Taking into account the age of the Airbus A300 fleet a time frame has been established for the operators activities, i.e. physical examination of the repairs, classification and assessment. The

evaluation time frame, see figure 2, gives priority for implementation to more sensitive structures.



<sup>1</sup> Priority for implementation to be given in older aircraft.

<sup>2</sup> Recommend classification to be done at time of survey if convenient.

Figure 2: Evaluation time frame

#### 3. PROCEDURE FOR REPAIR ANALYSIS

For the Airbus A300 the repair analysis consists of six major steps to categorize the repairs according to the categorization shown in chapter 2.3. For the categorization procedure, special repair questionnaires have been developed, e.g. for skin repairs, lap joint repairs etc. These repair questionnaires are based on a common understanding of major manufacturers and airlines. An example is given in Figure 3.

I - GENERALITIES OF REPAIR		II - GEOMETRIC DATA	
1- LOCATION Substructure: frame, rib, etc. Substructure: stringer, spar, etc. Panel number (last 5 digits)		To collect the necessary geometric data for the repair, indicate and sketch the following dimensions and data:	
2- COMPONENT IDENTIFICATION Substructure: skin, rib, frame, stringer, spar, etc. Repair: patch, doubler, etc.		3- CATEGORY "D" IDENTIFICATION A "YES" or "NO" answer is given according to the state of the repair in category D. If "YES", indicate the following: 1- REPAIR CONDITION 2- FASTENER TYPE 3- FASTENER PITCH (mm) 4- FIELD FASTENER 5- DOUBLER STRENGTH	
4- DESIGN ANALYSIS 1- Are there any additional repairs, rivets or existing joints within 10 mm? 2- Are there any existing repairs, rivets or existing joints within 10 mm? 3- Are there any existing repairs, rivets or existing joints within 10 mm? 4- Are there any existing repairs, rivets or existing joints within 10 mm? 5- Are there any existing repairs, rivets or existing joints within 10 mm?		5- DESIGN ANALYSIS 1- Are there any additional repairs, rivets or existing joints within 10 mm? 2- Are there any existing repairs, rivets or existing joints within 10 mm? 3- Are there any existing repairs, rivets or existing joints within 10 mm? 4- Are there any existing repairs, rivets or existing joints within 10 mm? 5- Are there any existing repairs, rivets or existing joints within 10 mm?	

Figure 3: Repair categorization questionnaire

To minimize the activities of the airlines the procedure of the repair analysis is built up in such a way, that each repair will be checked first for categories D and C. This avoids unnecessary

determination of an inspection program for time limited repairs.  
Figure 4 contains the procedure for repair analysis using the repair questionnaire.

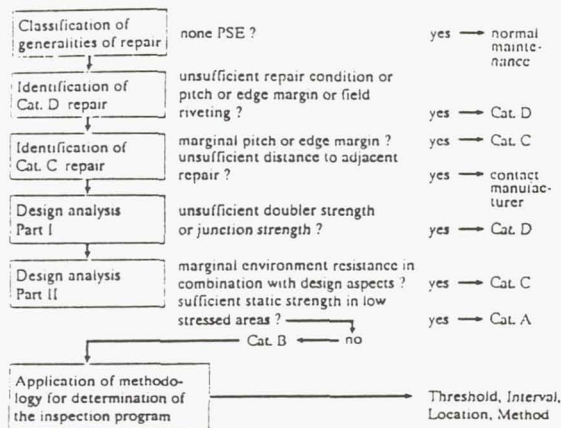


Figure 4: Procedure for repair analysis using the repair questionnaire

The first three steps of the procedure are easy to accomplish. Step 4, which is the design analysis Part I, includes a check of both, doubler strength and strength of fasteners. It is required that the doubler strength is greater than the strength of the skin and that sufficient fastener strength is available taking into account type of fastener, pitch, fastener diameter, skin/doubler thickness and location of repair.  
Step 5, which is the design analysis Part II, includes a check of the environmental conditions in combination with design aspects. The accomplishment of a grading leads to categories A, B or C.  
Step 6, to be performed for category B only, is the application of the methodologies for determination of the inspection program. These methodologies, which have to be applied to the majority of the repairs, are described in chapter 4.

#### 4. METHODOLOGY FOR DETERMINATION OF INSPECTION REQUIREMENTS FOR SKIN REPAIRS

For repairs categorized as B, an inspection program has to be determined. The methodologies to be applied on Airbus A300 aircraft skin repairs are described here. Furthermore, methodologies have been either finalized or are under development for repairs of the following Airbus A300 structure:

- fuselage longitudinal and circumferential joints
- skin of vertical stabilizer (metallic)
- door skin
- door surrounding frames
- door surrounding panels

Figure 5 shows a principle sketch of a skin repair.

In principle, four fatigue sensitive locations exist, for which the inspection requirements have to be determined:

- longitudinal rivet row on doubler adjacent to cut-out
- longitudinal rivet row on skin at doubler run-out
- △ circumferential rivet row on doubler adjacent to cut-out
- ▲ circumferential rivet row on skin at doubler run-out

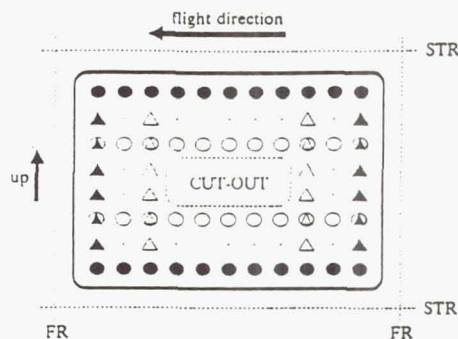


Figure 5: Principle sketch of skin repair

The inspection requirements to be determined are:

Threshold (TH) and Interval (I)

The determination of the inspection for the circumferential rivet rows is only required above the window line with one exception in the cockpit area.

For threshold determination two methods of different complexity have been developed according to the airlines requests. The detailed method leads to the maximum allowable threshold in contrast to the so-called simplified method which is less time consuming. For interval definition in general a detailed method is to be applied resulting in maximum allowable intervals for NDT inspection. For small repairs an alternative method is provided allowing detailed visual inspections. An overview about the application of the methods is shown in Figure 6.

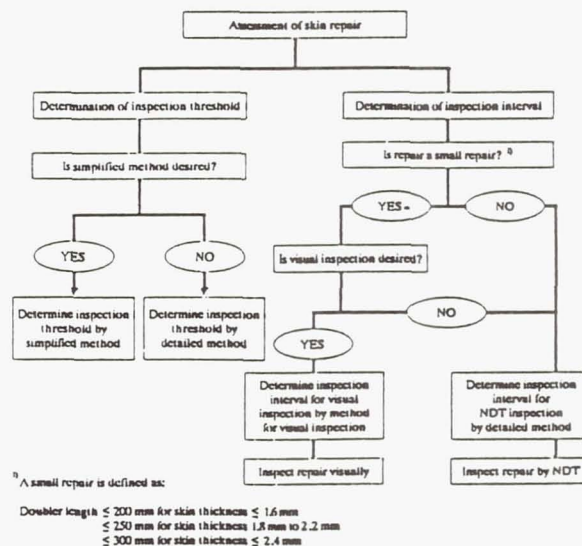


Figure 6: Alternative Methods for Determination of Inspection Program

#### 4.1 Determination of Inspection Threshold for Skin Repairs

##### 4.1.1 Detailed method

The methodology is based on the following procedure:



A so-called basic threshold is given for a specific skin repair for an unlimited cut-out in the upper shell of the rear fuselage at a defined location. All data for this repair as geometry, aircraft utilization, materials, rivet types etc. are exactly defined.

The inspection threshold for a repair at another location of the aircraft and/or different geometry and/or different other parameters can be determined by the operator by multiplying the basic threshold with factors considering the influence of the different parameters. The factors are to be determined from diagrams or tables which are supplied in the guidelines. At last the time of embodiment of the repair is to be added to the calculated threshold to obtain the inspection threshold in number of flights for the specific aircraft under investigation.

The equation for calculation of the inspection threshold of skin repairs is given below:

$$TH = TH_{basic} * LO * SC * DT * EC * RD * RT * RC * PI * ME * RR * UT * CO * AC * SR * SM * TR$$

with

TH<sub>basic</sub> : basic threshold  
 LO : location of repair  
 SC : size of cut-out - only for longitudinal rivet rows  
 DT : doubler thickness - only for doubler  
 EC : eccentricity  
 RD : rivet diameter  
 RT : rivet/bolt type  
 RC : former countersunk - only for skin  
 PI : rivet pitch  
 ME : edge margin - only for doubler  
 RR : number of rivet rows  
 UT : aircraft utilization - only for circumferential rivet rows  
 CO : countersunk depth  
 AC : countersunk depth/height  
 SR : distance of rivet rows  
 SM : distance of inner rivet rows - only for doubler  
 TR : time of embodiment of repair

The accuracy of the method is acceptable, considering, that all skin thicknesses are between 1.6 mm and 2.5 mm in the fatigue sensitive areas. Furthermore the maximum threshold is limited to the economic repair life plus time of embodiment of the repair.

The explanations given below are related to the longitudinal rivet rows only.

#### 4.1.1.1 Basic threshold for skin repairs

The basic threshold in flights is given in Table I.

TABLE I

BASIC INSPECTION THRESHOLD FOR SKIN REPAIRS

Item	TH <sub>basic</sub> (flights)	
	circumferential rivet rows	longitudinal rivet rows
skin	35 000 (▲)	30 000 (●)
doubler	45 000 (Δ)	38 000 (○)

The determination of the basic threshold is based on coupon specimens tested under constant amplitude loading at different levels. The specimens are shown in Figure 7.

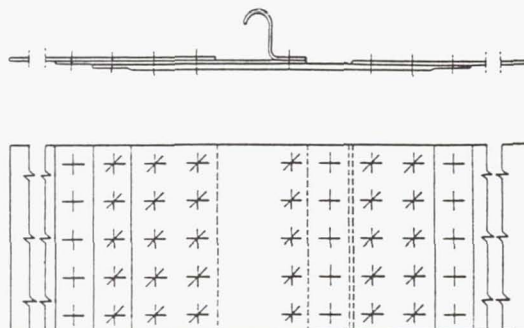


Figure 7: Specimens for coupon testing

The failure of the specimens occurred in both, skin and repair doubler, therefore the resulting SN-curve is valid for both locations. The applied aircraft spectrum for the longitudinal rivet rows is a one-step-spectrum due to internal pressure. The stress level used for calculation is based on analysis and considers in addition the results of former comparisons between coupon testing and full scale fatigue testing. Furthermore a scatter factor of 3 is used according to former agreement from French and German Airworthiness Authorities. The lower basic threshold for the skin compared with the doubler considers a countersunk in the skin to create a common basis for external doubler repairs and flush repairs.

#### 4.1.1.2 Factors applied on basic threshold

The major factors are explained in the following:

\* LO - location of repair

The factor LO, which considers the location of the installed repair, is based on the different stress levels in the fuselage. For the longitudinal rivet rows, which are loaded by internal pressure only, LO is determined using the hoop tension stress and the slope of the SN-curve used for the basic threshold determination. The location factors for the rear fuselage are shown in Figure 8.

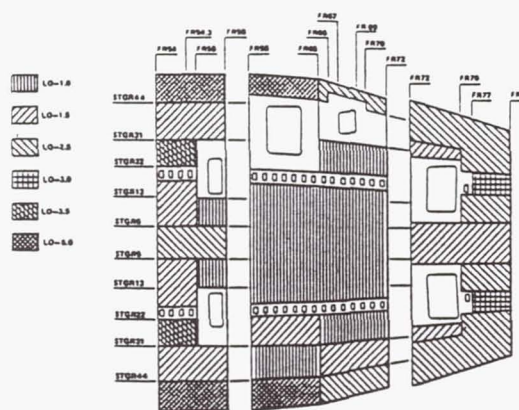


Figure 8: Location factors for longitudinal rivet rows of skin repairs in rear fuselage



\* SC - size of cut-out

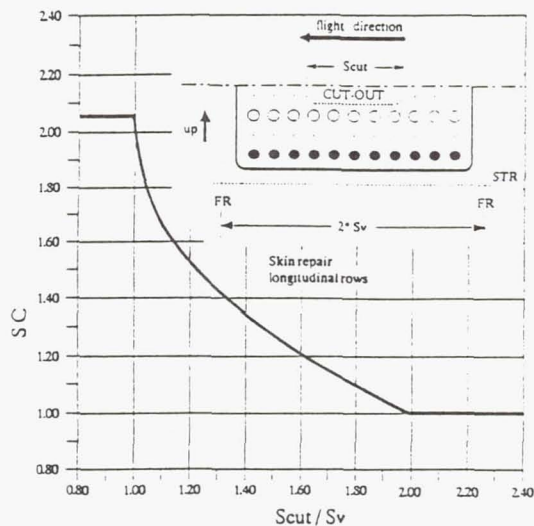


Figure 9: Influence of cut-out size

Figure 9 shows the effect of the cut-out size in relation to the length of a frame bay. This factor takes into account, that the load transfer is mainly a function of the cut-out size which is analyzed using Ref. /1/. The resulting stresses and the a.m. SN-curve are used to define the factor SC.

\* DT - doubler thickness

The influence of the doubler thickness is shown in Figure 10. DT is determined considering that the variation of the doubler thickness is not fully effective due to bending effects. Internal investigations led to the conclusion to consider half of the thickness variation for lap splices with thicknesses similar to Airbus A300 fuselage skin. The variation of stress is then converted into fatigue life variation using the slope of the relevant SN-curve.

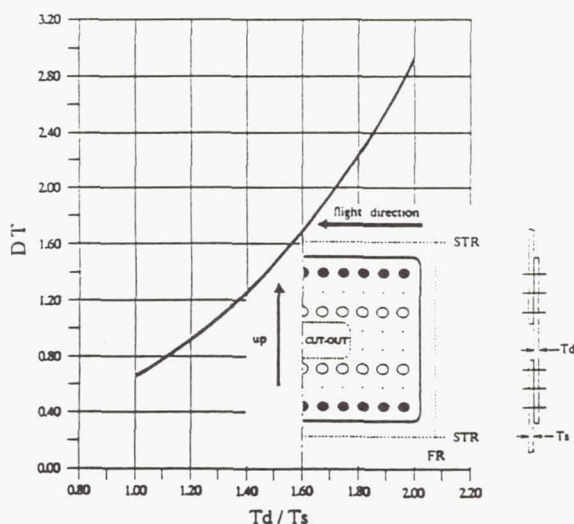


Figure 10: Influence of doubler thickness

\* RD - rivet diameter

The influence of the rivet diameter in relation to skin or doubler thickness is based on test results issued in Ref. /2/ and shown in Figure 11.

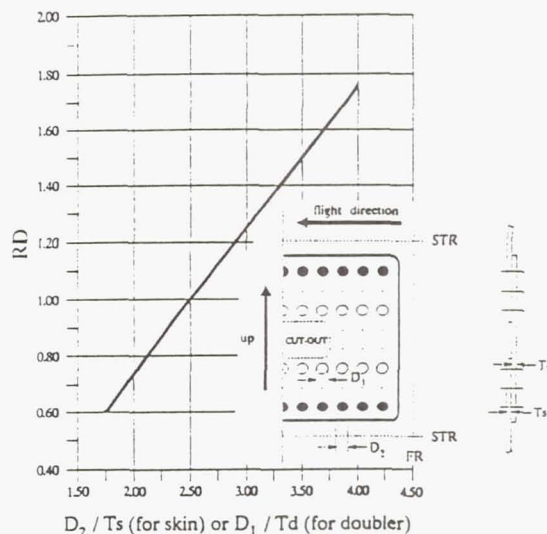


Figure 11: Influence of rivet diameter

\* RT - rivet/bolt type

The factor RT considers the various conditions of the rivets/bolts as heat treatment, bolt fit, shape of rivet head and material. The results given in Table II are an extract from the future content of the repair assessment program and are obtained from several coupon test series, which have been performed during the design of the various Airbus types.

TABLE II  
INFLUENCE OF RIVET/BOLT TYPE

Rivet type	RT	
	Manufactured head	Formed head or nut side
NAS 1097 DD	1.00 (c)	1.30 (p)
NAS 1097 D		
- heat treated	1.00 (c)	1.30 (p)
- non heat treated	0.50 (c)	0.65 (p)
Hi-Lok (Ti or Steel)		
- interference 1.0% D	1.30 (c)	1.69 (p)
- interference 0.3% D	1.10 (c)	1.43 (p)
- no interference	1.00 (c)	1.30 (p)

(c) ... countersunk head, (p) ... protruding head

Beside the major factors described above, ten additional factors have an influence on the fatigue life, i.e. on the inspection threshold of the repair. Most of these factors may be significant for repairs not in accordance with the recommendations defined in the SRM.

#### 4.1.2 Simplified method

The so-called simplified method for determination of the threshold is based on several assumptions about the conditions of the repair, i.e. about the

parameters influencing the threshold. These assumptions lead to certain values of the factors described in chapter 4.1.1, which are applied to the basic value  $TH_{basic}$  (see chapter 4.1.1). The inspection threshold is determined by the following equations:

- unlimited cut-out in all areas and limited cut-out above window line:

$$TH = 12\,000 * PI + TR \quad (\text{flights})$$

- limited cut-out below window line:

$$TH = 24\,000 * PI + TR \quad (\text{flights})$$

A limited cut-out is less than half a frame bay in longitudinal direction. For PI and TR see chapter 4.1.1.

For application of the simplified method some limitations have to be considered. The major limitations are:

- external doubler only
- skin thickness:  $T_s \geq 1.6\text{mm}$
- doubler thickness  $T_d$ :
  - unlimited cut-out
    - for  $1.6\text{mm} \leq T_s \leq 2.0\text{mm}$ :  $T_d = T_s + T$ ,  $T \geq 0.2\text{mm}$
    - for  $T_s > 2.0\text{mm}$ :  $T_d \geq T_s$
  - limited cut-out:  $T_d \geq T_s$
- solid fastener (no blind fastener)
- minimum three rivet rows

#### 4.2 Determination of Inspection Interval for Skin Repairs

##### 4.2.1 Detailed method for NDT inspection

The determination of the inspection interval is based on the same procedure as explained for the inspection threshold, i.e. a basic interval is to be multiplied by factors to consider all relevant deviations between the basic repair and the concerned repair. This method can be applied to all repairs independent on the size.

The equations and factors to be used are given below:

$$I = I_{basic} * LOI * SCI * MAI * DTI * PII * UTI$$

with:

$I_{basic}$ : basic interval  
 LOI: location of repair  
 SCI: size of cut-out - only for longitudinal rivet rows  
 MAI: material  
 DTI: doubler thickness - only for doubler  
 PII: rivet pitch  
 UTI: aircraft utilization - only for circumferential rivet rows

The maximum interval is limited to 12 000 to 18 000 flights depending on the Airbus A300 derivatives.

The explanations given in the following are again related to the longitudinal rivet rows of external skin repairs.

##### 4.2.1.1 Basic interval for skin repairs.

The basic interval in flights is given in Table III.

TABLE III

BASIC INSPECTION INTERVAL FOR SKIN REPAIRS

Item	IBasic (flights)			
	circumferential rivet rows		longitudinal rivet rows	
skin	4 500 (▲)	*1	6 000 (●)	*1
	5 000 (▲)	*2	7 000 (●)	*2
doubler	9 000 (Δ)	*3	12 000 (○)	*3

NDT methodology for external repairs:

- \*1 ultra-sonic from outside
- \*2 eddy-current high frequency from inside
- \*3 eddy-current high frequency from outside

The determination of the basic interval for the longitudinal rivet rows is based on the results of a multiple crack pattern evolution in a two-rivet row single shear joint of a large test article. This crack pattern is considered to be more realistic than assuming arbitrary crack pattern consisting of one 0.05 inch flaw at a center fastener hole and several 0.005 inch flaws at adjacent fastener holes. The inspection interval is calculated by application of a scatter factor of 2. Furthermore alternative inspection methods requiring different access have been considered. The detectable crack lengths are based on a detection probability of 95 percent and a confidence level of 90 percent.

##### 4.2.1.2 Factors applied on basic interval

For the interval of the longitudinal rivet rows five factors are of special importance, i.e. LOI, SCI, MAI, DTI and PII. In principle the factors LOI, SCI, and DTI have a similar effect as the corresponding factors LO, SC and DT on the threshold as explained in chapter 4.1.

Details of the remaining two factors are explained below.

##### \* MAI - material

In the majority of the areas, the skin and doubler material is 2024T3/T42 resulting in a material factor of 1.0. In exceptional areas the material 7075T6 is used which leads to an average material factor of 0.7 considering the increased crack growth in this material.

##### \* PII - rivet pitch

Figure 12 shows the influence of the rivet pitch on the inspection interval. The factor PII is the result of a comparative crack propagation calculation showing increasing crack propagation periods for increased rivet pitches. This effect is in line with results given in Ref. /3/.

Additionally, Figure 12 shows the effect of the pitch on the threshold, which has a reverse effect as verified in many investigations.

For the circumferential rivet rows the factor UTI has to be applied to consider the concerned Airbus A300 derivative, i.e. B2, B4, C4 and F4, and the average airborne flight time. The factor UTI is significant for the following conditions: A300B4 derivative, rear fuselage and medium/long range utilization. For example a six hour flight leads to a reduction of the interval of 33 percent compared with a one hour flight of a B2 aircraft.



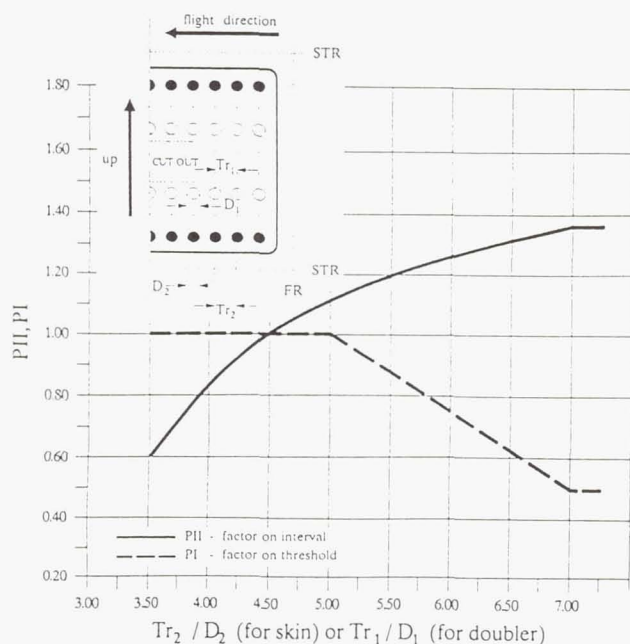


Figure 12: Influence of rivet pitch

#### 4.2.2 Visual inspection of small skin repairs

For easily accessible areas of the fuselage, i.e. below the window line, an alternative method is provided which allows detailed visual inspections. This method is based on the assumption, that the skin in the doubler area or the doubler can completely crack without immediate impact on the safety. Assuming a detectable crack length of 25 mm at each side of the doubler the initial crack length for a crack propagation calculation is the doubler length plus 50 mm. Crack propagation calculations were performed for different skin thicknesses, i.e. for different stress levels. A scatter factor of 2 is applied on the crack propagation period between detectable and critical crack length, as done for the interval for NDT inspection as well.

To reach reasonable inspection intervals and to limit the possible crack length in the skin the visual inspection is limited to so-called small repairs. Small repairs are defined by the doubler length  $S_d$  depending on the skin thickness  $T_s$  as follows:

- for  $T_s \leq 1.6 \text{ mm}$ :  $S_d \leq 200 \text{ mm}$
- for  $1.8 \text{ mm} \leq T_s \leq 2.2 \text{ mm}$ :  $S_d \leq 250 \text{ mm}$
- for  $T_s > 2.2 \text{ mm}$ :  $S_d \leq 300 \text{ mm}$

Figure 13 shows the intervals for visual inspections depending on the doubler length and the skin thickness. The intervals are between approximately 300 flights for a 200 mm doubler on the 1.6 mm thick skin and approximately 12 000 flights for a 125 mm doubler on a 2.4 mm thick skin. If the interval is not convenient considering the airline's maintenance schedule the interval for NDT inspection is to be determined acc. chapter 4.1.1.

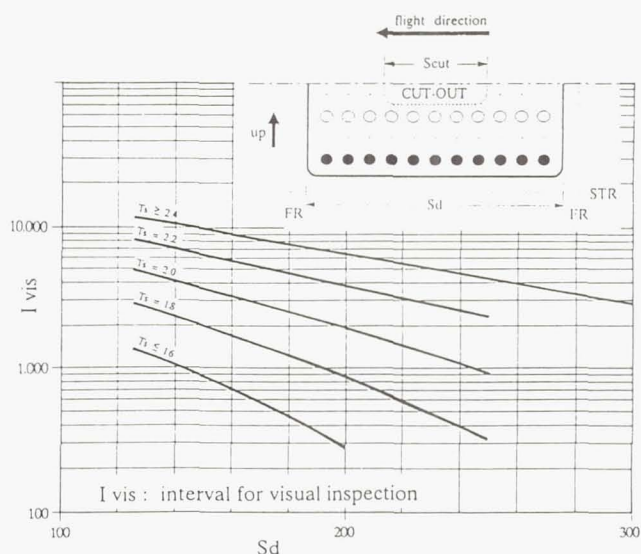


Figure 13: Intervals for visual inspection of small repairs

#### 5. CONCLUSION

The developed repair assessment program is considered to be an adequate response to the requirements defined by the AATF steering committee. The main subject of the repair assessment program is the methodologies for determination of the inspection program. A draft of the detailed methodology for assessing skin repairs has been presented to the airworthiness authorities, members of the AATF and Airbus A300 operators. The method, which was tailored to the airlines' requirements regarding limitation of work load and complexity, has been accepted.

The presented methodologies have significant advantages for the airlines, since they apply to nearly all fuselage skin repairs, even to those not in accordance with the manufacturer's instructions. The detailed methodology allows detailed evaluation with simple calculations and provides maximum allowable thresholds and intervals. Furthermore the methodology contains guidance material for design of repairs with high inspection thresholds and long inspection intervals. After finalization of the methodologies for other fuselage structural elements prone to repairs (early 1992) and the repair assessment document, the airlines will be in a position to assess more than 90 percent of the existing fuselage repairs without support of Airbus Industrie.

#### 6. REFERENCES

- /1/ Flügge, W. "Stress Problems in Pressurized Cabins", NACA TN2612
- /2/ Hertel, H. "Ermüdungsfestigkeit der Konstruktionen" Springer-Verlag Berlin/Heidelberg, 1969, p. 443
- /3/ Swift, T. "Repairs to Damage Tolerant Aircraft", International Symposium on Structural Integrity of Aging Airplanes, Atlanta/Georgia, March 1990



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# COMMUNICATION

## AN IMPORTANT ELEMENT OF MAINTENANCE AND REPAIR

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**People from the airlines, the FAA, and the manufacturers have worked long hours in an effort to improve the Service Difficulty Reporting system. Their work to date is summarized as follows.**

- **Goals were generalized into three broad work statements;**
  - (a) **Design a worldwide reporting system to provide safety alerts to aircraft operators, manufacturers, repair facilities and regulatory authorities.**
  - (b) **Design a companion system to provide worldwide reliability experience.**
  - (c) **Overhaul regulatory requirements to be consistent with (a) and (b) to provide information necessary and useful for public consumption.**
- **Focus soon narrowed to the specific need to improve the SDR system. Three subcommittees were formed;**
  - 1) **Data Collection Subcommittee**
  - 2) **Data Analysis Subcommittee**
  - 3) **Safety/Reliability Action Subcommittee**
- **Data Collection Subcommittee was assigned 3 (short-term) goals;**
  - 1) **Improve reporting by air carriers by achieving more consistent and complete reporting, giving consideration to aircraft age. Submit these reports electronically into the FAA database.**
  - 2) **Improve repair station reporting.**
  - 3) **Identify the information that should be reported, giving consideration to the current regulation and recommend changes as appropriate.**

Recent major accidents of US-registered aircraft have given rise to calls for improved systems to detect incipient problems before they become accidents. Pressure has been increased by the introduction of bill HR 3555 which formally calls for the setting up of a safety information system.

Both the FAA and the Air Transport Association recognized the need to improve airworthiness communications and initially separate activities led (in 1989) to the establishment of a joint FAA/Industry committee, the Improved Airworthiness Communications Steering Committee (IACSC), having the proposed objective of improving the reporting and analysis of safety-related operational data.

- Initially, the IACSC was a committee reporting to the Airworthiness Assurance Task Force.
- The Committee recognized the need to improve communication in the aircraft maintenance environment.
- To this end, Committee goals initially included:

Gather meaningful data.

Develop a common reporting format.

Ensure Foreign Air Carrier participation, a worldwide database.

Assure a worldwide alerting system.

Ensure consistent data collection/reporting.

Encourage electronic-based data handling, including electronic data submission and distribution.

Define consistent reporting requirements within ADs; corrosion findings; SSID findings.

- **PURPOSE OF THE S D R SYSTEM**

#### **121.703 MECHANICAL RELIABILITY REPORTS (SERVICE DIFFICULTY REPORTS)**

Enhance air carrier safety by:

1. Gathering data that identifies mechanical failures, malfunctions, and defects that reasonably may be expected to cause a serious hazard to the operation of an aircraft.



**AND**

2. **Developing and implementing corrective action to eliminate these failures, malfunctions, and defects once they have been identified as hazards to safe aircraft operations.**

The Government Accounting Office (GAO) reviewed the SDR system and identified a number of problem areas which include:

**Problem:**

1. **SDR reporting from airline to airline is inconsistent. The database lacks credibility.**

**From GAO Report:**

"The chief reason that FAA engineers and airline personnel are dissatisfied with the SDR program is that its data are of low quality."

"....airline officials attribute reporting differences to vague reporting requirements, leading to varying interpretations of what should be reported and to airlines' concerns over the public's access to malfunction reports in accordance with the Freedom of Information Act. Concerned about public disclosure of SDR data, some airlines are reluctant to submit malfunction reports to FAA."

**Recommendations:**

- a. **That 121.703 be amended to clarify existing regulation with the intent of reducing interpretation as to what is required to be reported. Clear and concise regulation reduces interpretation and enhances the consistency of the reporting required by this regulation.**
- b. **That the aircraft structural reporting requirements of existing FAR 121.703, together with provisions for the reporting of factors relevant to aging aircraft and corrosion protection be gathered into a new regulation, FAR 121.704.**
- c. **That airlines enter SDRs into the FAA's database electronically. A standardized data entry format with menus and built-in edits enhances consistency.**

- d. That certain exemptions from the Freedom of Information Act (FOIA) be made applicable to the air carrier information submitted per the requirements of these regulations. This protection (from FOIA) would encourage the willful reporting of safety related information into these databases.

**Problem:**

2. SDR data is not timely. Weeks may elapse between an airline reporting a problem and the problem being data-based and distributed by the FAA.

**From GAO Report:**

"The SDR program's lack of timeliness is a critical flaw in its ability to effectively serve its users."

"The SDR program does not contribute to the timely correction of conditions affecting aircraft safety because, aside from storing data on a computer, it is a paper-based, manual process."

"From beginning to end, a single malfunction report will spend approximately 6 weeks in the SDR processing system before becoming available to analysts."

**Recommendations:**

- a. That FAA encourage airlines to enter SDRs into the FAA's database electronically. (FedEx and Northwest have been doing so in a test program.) The shift from paper to electronic submission of data to the FAA should provide a database that is almost real-time.
- b. That FAA make the SDR database available to the certificate holders, repair stations, and manufacturers for electronic data retrieval capability. The database would provide real-time history to participating parties and become an information resource. In other words, participation would be enhanced because the system would provide an information benefit to the user.

**Problem:**

3. The data is not analyzed. Problems are not identified, corrective actions are not developed or implemented.

**From GAO Report:**

"...FAA engineers....told us that poor data quality discouraged them from analyzing SDR data."

**Recommendations:**

- a. Same as problem 1. These recommendations, if implemented, should improve database accuracy and credibility.

**NOTE:** Industry and the FAA are participating in on-going activities (Improved Airworthiness Communications Committee's Data Analysis & Action Subcommittees - now known as the International Airworthiness Communications Working Group) to improve problem recognition and action implementation.

- The subcommittee further recommended that;
  - 121.703 be renamed Operational Difficulty Reports and focus on the reporting of operational difficulties.
  - 121.704 be created to capture Structural Difficulty Reports.
  - The word "flight" be eliminated from several areas of the regulation. Significant safety-related events occur while the aircraft is on the ground.
  - Reports should focus on A/C serial number rather than registration number. Registration numbers can change rapidly in today's environment, serial numbers do not.
  - Reports capture aircraft hours and cycles to address aging aircraft concerns. Also gather information that details the location of structural difficulties.
  - The system be designed to code reports to the discrepancy. Focus on alerting to the discrepancy rather than on the corrective action (fix). Oftentimes, the fix isn't!
  - Component problems be identified by sorting by the part number rather than coding the discrepancy to the part number level.



- SDR's incorporate a "turn-off" provision that curtails reporting once a problem is known, an AD or SB has been issued, and the repair can be affected without deviating from published instructions. This eliminates filling the database with reports of known problems.

AD or SB effectiveness can be captured by issuing reporting instructions within these documents. Don't use the SDR system to monitor these corrective actions.

- 121.705 (Mechanical Interruption Summary) reporting be eliminated. These reports are reliability, rather than safety related. 121.373 requires Surveillance and Analysis; therefore, the monitoring of these events can take place within the operators Reliability Program.
- Recently, the IACSC became a working group reporting to the Air Carrier/General Aviation Maintenance Subcommittee. This subcommittee, in turn, reports to the Aviation Rulemaking Advisory Committee.

These proposals are currently in the preliminary stages of the NPRM process.

- Examples of 121.703 modification:

- (1) Fires during flight and whether the related fire-warning system functioned properly;
- (2) Fires during flight not protected by a related fire-warning system;
- (3) False fire warning during flight;

Replaced by

- (1) All fires and, when monitored by a related fire-warning system, whether the fire-warning system functioned properly. All false fire or smoke warnings that require the use of emergency procedures;

- (6) Engine shutdown during flight because of flameout;
- (7) Engine shutdown during flight when external damage to the engine or airplane structure occurs;
- (8) Engine shutdown during flight due to foreign object ingestion or icing;
- (9) Engine shutdown during flight of more than one engine;

Replaced by

- (3) Any engine flameout during taxi or engine flameout or shutdown during flight. Intentional engine shutdowns for flight crew training or during test flights need not be reported;

**Let me expand on 121.704 just a little.**

The work of the IACSC has led to the formulation of recommendations for a proposal of a new FAR, FAR 121.704, which specifically addresses defects in aircraft structures and the problems of aging aircraft. Based on this recognized need to insure the continued airworthiness of aging aircraft, this rule will capture structural defects that could have safety-of-flight implications. The reporting requirements of 121.703 (a) 14 & 15 have been revised and incorporated in this proposed FAR. The required reporting focuses on discrepancies discovered in primary structure and principle structural elements. Furthermore, it will capture discrepancies discovered in composite materials when they comprise primary structure or principle structural elements. This proposal also incorporates a provision that provides relief from additional reporting once a structural problem has been recognized (by the issuance of an A. D. or a manufacturer's service bulletin) and a published corrective action can be followed to affect a repair. This provision reduces the economic burden to air carriers in that it eliminates the requirement to continue reporting known problems. The continued reporting of known problems that have resulted in the issuance of an A. D. can be accomplished by placing reporting requirements within the A. D.

**The final goal of the Data Collection Subcommittee....Address Repair Station reporting.**

The subcommittee's recommendation to improve repair station reporting is to clarify 145.63. 145.63 should be modified to state that any item discovered on a Part 121 Air Carrier's aircraft that must be reported per the requirements of 121.703 or 121.704 will be submitted by the repair station to the FAA and a copy of the report forwarded to the 121 certificate holder. The certificate holder's responsibility for reporting these occurrences as required per 121.703 or 121.704 will be delegated to the repair station. Since the items that are reportable under these sections can have safety implications, timely reporting is most likely to occur when the repair station submits the discrepancies it discovers. Additional rationale for this recommendation is the realization that duplicate reporting occurs because of duplicate requirements. The repair station currently submits a report to the FAA as required per 145.63 and the air carrier submits the same report to the FAA as required per 121.703/121.704. These changes are recommended to insure more timely reporting and to eliminate duplicate reporting.

- **What remains to be done?**

Industry and government need to continue working together (International Airworthiness Communications Working Group) to determine best methods of analysis, alerting, and action implementation.

Determine who should be responsible for gathering and analyzing the data, the FAA or the OEM.

There are problems and benefits associated with both alternatives.

At the OEM; (Problem) corporate liability concerns .... (Benefits) solves Foreign Air carrier participation & FOIA Concerns.

At the FAA; (Problem) Foreign Air carrier participation, FOIA Concerns.

Determine the best SDR data distribution alternatives; Intelligent Service Network, On-line access, CD-ROM distribution.

We need to bring together the various structural data gathering and reporting activities.

What follows is a proposal of a method to implement structural data collection and analysis in a databased mapping environment. This system has been developed by one European Air Carrier. In addition to databasing the relevant defect and repair information, it displays that information in a 3-D format. Repairs can then be recalled and displayed as a picture. This Computer Aided Design (CAD) software enables the user to visualize repairs as they occur on an aircraft or a fleet of aircraft. Its implementation could be a tremendous aid in structural defect detection and analysis.



## **Present and Future Aircraft Structural Condition Data Collection and Analysis Activities**

- 121.703 Mechanical Reliability Reports (contain structural reports)
- 121.704 Structural Difficulty Reports ?
- Repair Assessment Reporting ?
- S. I. D. Reporting
- Corrosion Prevention Reporting
- Aging A/C Service Bulletin Inspection Findings Reporting



**TODAY**

**Different Reports - Different Forms  
Different Databases - Different Destinations**



**A PROPOSAL FOR TOMORROW**

**Begin Work To Develop A Common Reporting  
System That Combines These Activities**



**GOAL**

**One Form - One Report - One Destination - One Database**

**RESULTING IN**

**A TOTAL PICTURE OF A/C STRUCTURAL CONDITION**

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## Human Factors in Aircraft Maintenance and Inspection

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### Summary

The events which have led to the intensive study of aircraft structural problems have contributed in no less measure to study of the human factors which influence aircraft maintenance and inspection. Initial research emphasis on aging aircraft maintenance and inspection has since broadened to include all aircraft types, both new and old. The role of today's aircraft maintenance workforce is indeed a complex one as it is strongly influenced by the need to maintain the broad technology mix embodied in the current fleet. Technicians must be equally adept at diagnosing and repairing "old-tech" 727, DC-9 type aircraft and "new-tech" 767, and MD-11's. Their skills must include ability to repair sheet metal and composite materials; control cables and fly-by-wire systems; round dials and glass cockpits. Their work performance is heavily influenced by others such as designers, technical writers, job card authors, schedulers and trainers. This paper describes activities concerning aircraft maintenance and inspection human factors.

In the past few years a number of very prominent air carrier accidents have highlighted the role of aircraft maintenance in aviation safety. The Aloha Airlines 737 accident and the United DC-10 accident at Sioux City, Iowa are two relatively recent events which were precipitated by maintenance and inspection problems. In response to these and earlier events the FAA's Office of Aviation Medicine has been assigned the task of investigating the role of human performance in the maintenance and inspection of air carrier aircraft. The FAA has developed a research program of study in several areas. These include Training, Human Error in Inspection, Work Environment, Job Performance Aids, Organizational Factors and Communication including study of hypermedia applications to enhancing information flow and accessibility. In addition a number of parallel activities are being conducted in this program including presentation of conferences dealing with several different human factors topics, and production of a Human Factors Guide for maintenance. This latter publication will present human factors information in a form usable by maintenance managers, designers, and FAA oversight personnel.



The research being accomplished in this program falls under the National Plan for Aviation Human Factors. The National Plan is focused primarily on research activities and merges the work of FAA, NASA and some elements of the Department of Defense. The Plan was developed with input from these and other government organizations, academia and private industry. The purposes of the National Plan are four-fold viz:

1. Identify significant human performance issues
2. Direct scarce resources to highest payoff
3. Transfer human factors technology
4. Communicate research needs

Scientific Task Planning Groups (STPG) met on several occasions to identify problems and formulate proposed research activity to address these problems. The STPG's covered four broad topics: Flight Deck, Air Traffic Control, Flightdeck - ATC Integration, and Maintenance. This last topical area covers both aircraft maintenance and airways facilities maintenance. This paper considers only aircraft maintenance.

The FAA maintenance human factors program addresses three sub-topics as specified by the National plan; a) Personnel and Training, b) Advanced Technology Systems, and c) Environment and Organization. The primary goal of this work is the enhancement of the performance of the human as a key element in the aviation maintenance system. Several activities or tasks within these sub-topics are described as follows.

#### Activity 1: Advanced Technology for Maintenance Training

This task deals with the development of advanced computer-based instruction systems. These new systems are called Intelligent Tutoring Systems (ITS). They embody more sophisticated software than traditional computer-based instructional systems. ITS software includes subject matter expert and instructor modules which are capable of adapting to individual student capabilities and needs. A prototype ITS for maintenance training has been developed. Future activity will concentrate on using and evaluating this ITS in realistic training settings.

#### Activity 2: Study of Human Error in Inspection

This activity involves laboratory and field study of factors which affect inspection accuracy and reliability. Particular attention will be given to visual inspection and non-destructive inspection (NDI) equipment and its use. Laboratory experiments have been devised to permit controlled study of factors which affect detection of various types of flaws or defects.

### Activity 3: Human Factors Guide for Maintenance and Inspection

This document will be a compendium of human factors information useful to maintenance managers, designers, and FAA oversight personnel. It will consist of information obtained from the current research program, (as described under other activities listed in this paper) and information from other sources. For example there is a very large literature on vigilance. This information base will be reviewed and material pertinent to the aircraft maintenance case will be extracted and appropriately structured to provide useful information to those with special interests in aircraft maintenance.

### Activity 4: Human Factors Conferences and Workshops

To date, five such conferences have been conducted. The purpose of these meetings is to bring together human factors experts, operational personnel, government representatives, airframe and system manufacturers, union representatives and others to discuss issues that are pertinent to maintenance human factors. These conferences facilitate communication among people with various viewpoints and expertise and help with problem solution by exchanging knowledge and experience. Conferences have been held on such topics as Training, Information Exchange and Communication, Work Environment and others. It is expected that these conferences will continue for the foreseeable future. The next planned conference will be held in January 1992 in the Washington, D.C. area with the topic "Design for Maintainability."

### Activity 5: Guidelines for Managing Communication

This activity focuses on development of guidelines for effective communication in maintenance organizations. Optimal methods for assuring two-way information flow will be developed and described as well as identification of organizational factors which facilitate communication.

### Activity 6: Job Performance Aids for Maintenance

Many industries have developed job performance aids for their particular activities. Some of these may be appropriated or modified for use in the air carrier maintenance industry. This task will examine several existing aids and select one for further development. After appropriate development, the selected system will undergo evaluation with a volunteer airline or airframe manufacturer.

#### Activity 7: Use of Hypermedia in Aviation Maintenance

A prototype system will be developed to determine the applicability of hypertext technology to aviation maintenance documentation. Hypertext technology will apply artificial intelligence concepts to such documentation as aircraft maintenance manuals. Appropriately applied, this technology could greatly reduce the time technicians spend in searching documents for needed information and could also contribute to reduction of maintenance error.

#### Activity 8: Study of Maintenance Work Environments

Work conditions can certainly influence performance of maintenance tasks. Such factors as ambient temperature and lighting, work site access and others can affect maintenance quality with a consequent effect on safety. This task will develop guidance on appropriate working conditions based on existing information for other industries and on information developed during on-site evaluations of air carrier maintenance work sites.

The study of aircraft maintenance human factors will continue for the foreseeable future. As the research progresses, additional topics in need of study have become apparent such as application of crew resource management (CRM) principals to aircraft maintenance. The program will make appropriate adjustments to assure that these additional topics are accounted for. Also those original tasks, as described in this paper, will continue to receive appropriate emphasis and directional changes if need be to provide maximum information to constituent groups.



# Using Intelligent Simulation to Enhance Human Performance in Aircraft Maintenance <sup>1</sup>

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## ABSTRACT

Human Factors research and development investigates the capabilities and limitations of the human within a system. Of the many variables affecting human performance in the aviation maintenance system, training is among the most important. The advent of advanced technology hardware and software has created intelligent training simulations. This paper describes one advanced technology training system under development for the Federal Aviation Administration.

## 1.0 INTRODUCTION/BACKGROUND

A myriad of human factors challenges associated with aircraft maintenance exist. Maintenance and inspection are essential to ensure continuing airworthiness on the aging fleet. New advanced technology procedures for aircraft further complicate the aviation maintenance system. The human is a critical component of the maintenance system. Therefore, research and development must address a variety of issues affecting the human in maintenance. Figure 1 depicts a few of the factors that can affect human performance in the maintenance system.

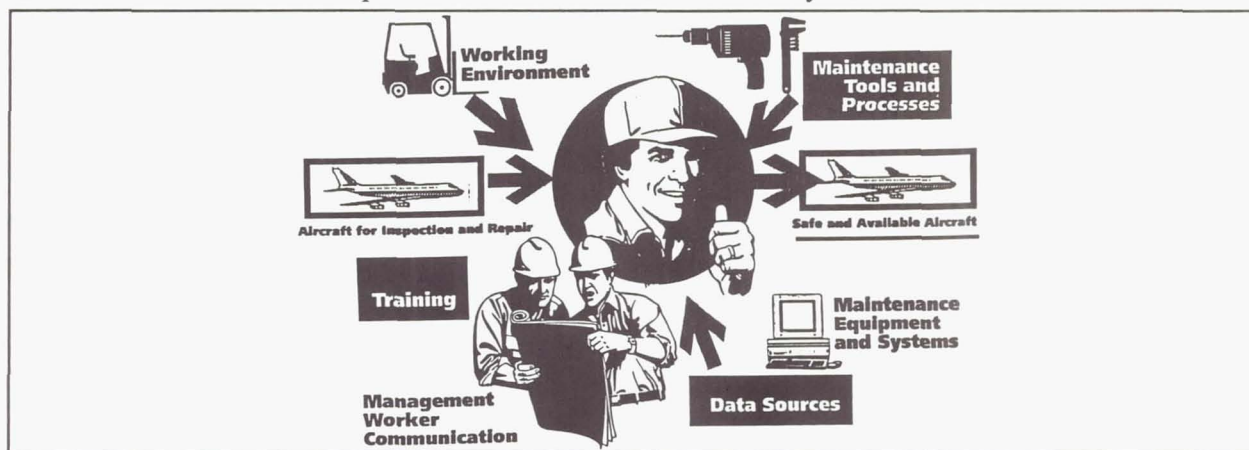


Figure 1. The Human in the Maintenance System

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The "work environment" includes not only physical characteristics such as noise, temperature, and illumination levels, but also less tangible measures associated with management-labor communications, corporate goal setting and attainment, and individual decision making by maintenance personnel. "Data sources" include the variety of technical information that a technician must access to perform and record maintenance actions. "Training" includes the many activities that prepare the human to perform a given maintenance task. Training must also help the human maintain an appropriate level of technical proficiency for safe, efficient, and effective job performance.

Human factors research and development is recognized as a critical component of aviation safety. *The Aviation Safety Act of 1988* (PL100-591) (ref.1) mandates that research attention be devoted to a variety of human performance issues including aircraft maintenance and inspection. Similarly, *The Aviation Security Improvement Act of 1990* (PL101-604) (ref. 2) states that the FAA "shall review issues relating to human performance in the aviation security system with the goal of maximizing such performance." *The National Plan for Aviation Human Factors* (ref. 3) is but another official recognition that the human is a most critical factor in aviation safety.

The Biomedical and Behavioral Sciences Division of the FAA Office of Aviation Medicine is engaged in a research program entitled "Human Factors in Aviation Maintenance." The program is conducting research in the areas listed in Table 1.

- 
- Advanced Technology Training Systems
  - Advanced Technology Information Systems
  - Analysis of Human Error in Maintenance and Inspection
  - Communications in Maintenance Environments
  - Development of Human Factors Guide for Maintenance
  - Human Factors Work Site Evaluations
- 

**Table 1.** Human Factors in Aviation Maintenance Research Topics

The comprehensive research program is described elsewhere (ref. 4). This paper will describe only the research and development task associated with advanced technology training systems.

## **2.0 TRAINING TECHNOLOGY**

Demographic data (ref. 5) indicates that there will be a distinct shortage of qualified aviation maintenance technicians (AMTs) by the year 2000. The obvious solution is to create methods that permit less qualified personnel to conduct maintenance and inspection by using advanced technology job aiding equipment (i.e., make the tools smarter and the job easier). A second way would be to devise new methods that will train, efficiently and effectively, more personnel as AMTs (i.e., make the human smarter). Of course, a hybrid of these two approaches would be to create job aids that also train (i.e., smarter tools and smarter people).



The suggested hybrid approach requires research and development on the use of artificial intelligence technology for both training and job aiding systems. The research must address issues associated with interface design and software optimization for portable computer systems. The advanced technology training system is called an "intelligent simulation". It is "intelligent" because it contains software components that provide extensive modeling and feedback that approach what might be provided by a human simulation instructor.

The advanced technology training project reports the current status of maintenance training, develops a prototype training system, and evaluates the system in maintenance training environments (ref. 6). The training is described in the following sections.

### 3.0 Environmental Control System (ECS) Trainer

The ECS Trainer (Figure 2) permits the student to interact with a simulation of the Boeing 767-300 air conditioning portion of the ECS. The system is designed to train for troubleshooting - the most complex of maintenance tasks. The student can operate the system under normal or malfunction conditions.

To troubleshoot the malfunction, the student looks at displays, manipulates controls and switches, and tests, inspects and replaces components. While troubleshooting, the student has access to operational information for the equipment. The student may also ask for advice on how to solve the problem most effectively.

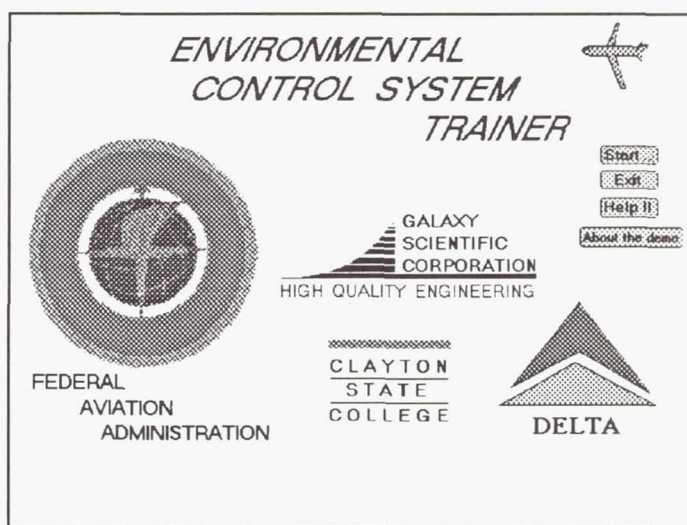


Figure 2. Environmental Control System Trainer

The following sections of the paper will familiarize the reader with the ECS Trainer from several different viewpoints. First, the interface design is discussed. Then, the general instructional design is addressed. Finally, the implications and possible future directions of the system are discussed.



### 3.1 Interface Design

Figure 3 shows the primary (Overview) display used by the student. The system was designed so that the majority of relevant troubleshooting information is only one display away from the primary display. This design prevents the student from getting "lost" while in the Trainer.

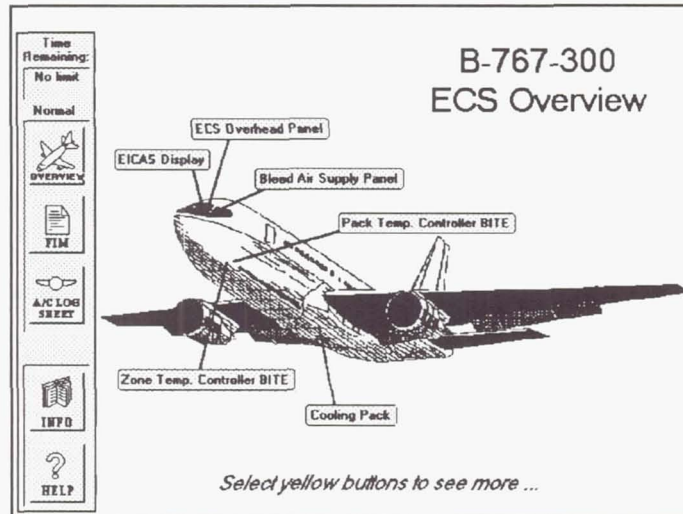


Figure 3. Primary ECS Display

From the Primary ECS display the student may access the displays, controls, and components that are required to troubleshoot a malfunction. For example, if the student selects the overhead panel there are two modes of operation. The first mode, shown in Figure 4, will provide basic instruction on how the equipment operates. The second mode, shown in Figure 5, provides a simulation of the system which the student can use for real-time operational practice. An expert system, embedded in the simulation, advises the student on system operation as needed or requested.

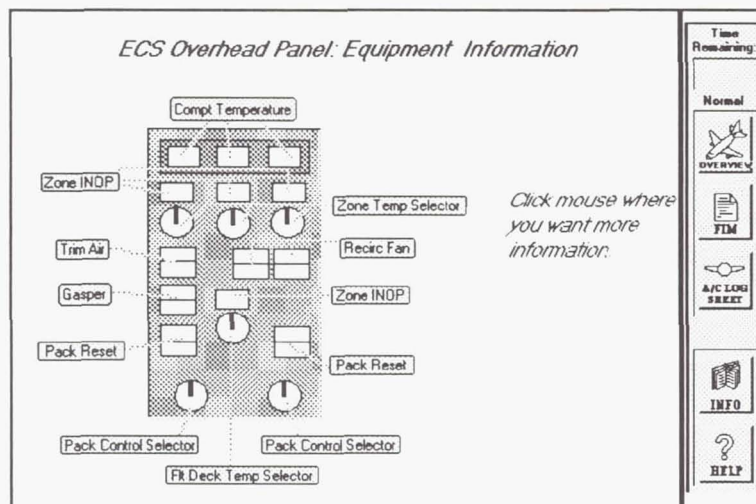


Figure 4. Equipment Information

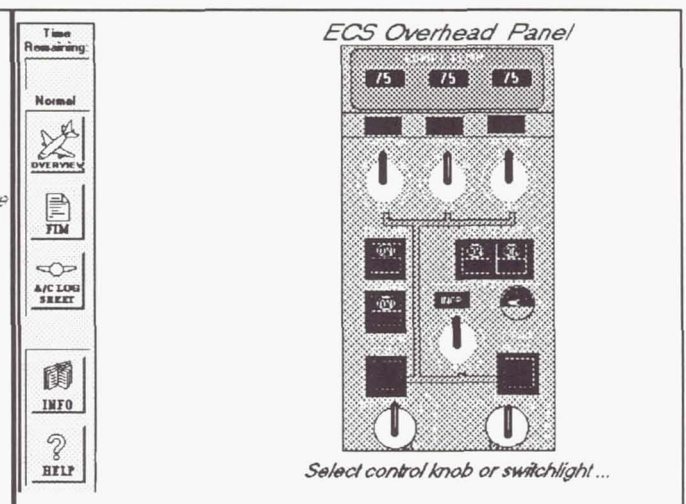


Figure 5. Equipment Operation

From the Primary ECS display the student can access the Fault Isolation Manual (FIM). The FIM, shown in Figure 6, is taken directly from the airline's training documentation. In the training system the FIM becomes a dynamic troubleshooting display that provides advice and feedback.

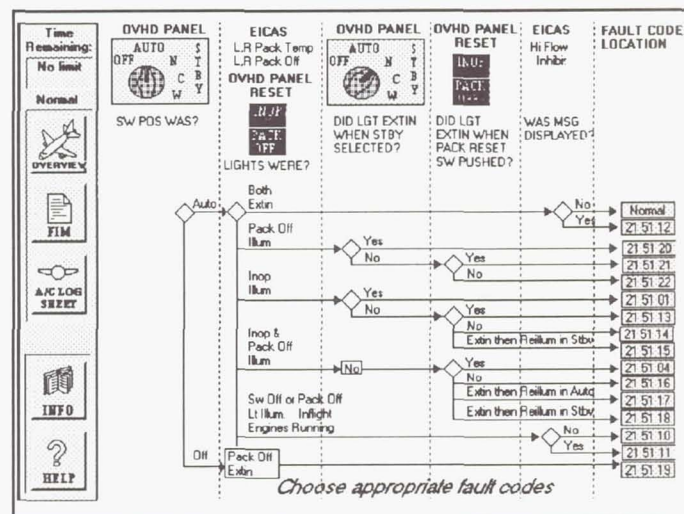


Figure 6. Page from Fault Isolation Manual

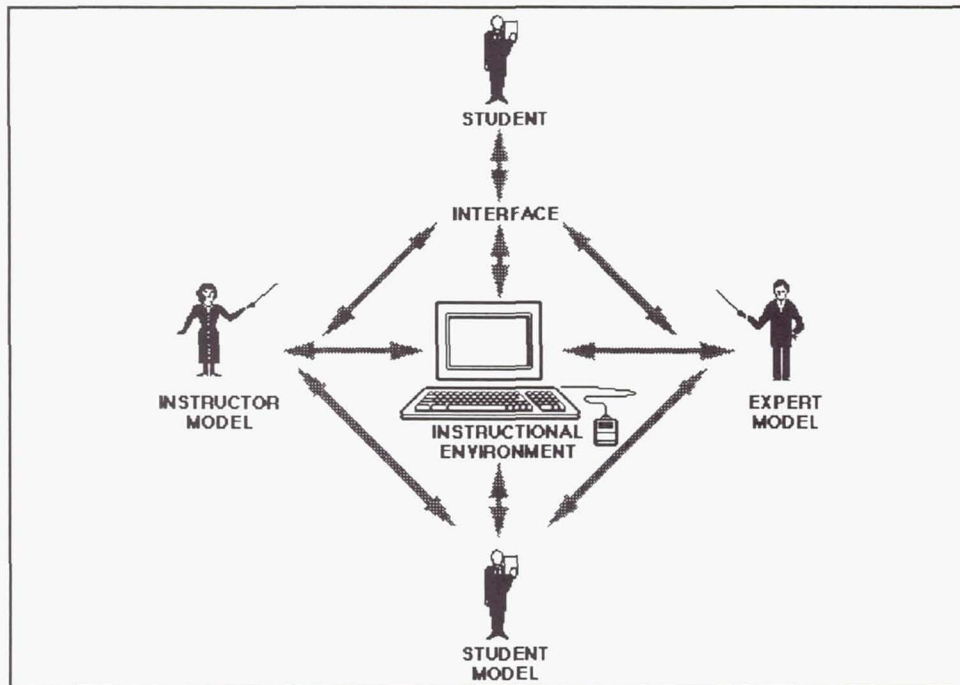
The student can also select "Help" from the Primary ECS display. The "Help" button provides three kinds of assistance. Help on the "current screen" describes how to use the features that are specific to the current display. The "advice" option provides hints to the student on how to troubleshoot a malfunction. The advice is based on the information contained in the FIM. The "how to use the system" option describes how to use the features that are common to all displays.

### 3.2 Instructional Design

The ECS intelligent simulation is designed with the instructional philosophy that "adult learners learn best by doing". The instructional design, therefore, permits the learner to operate the equipment in any way. The student can ask for help and advice at any time and is provided advice when it is badly needed. The intelligent simulation does not force the learner to follow any specific order. Essentially, the system adapts to the needs and learning characteristics of the student. This type of training system is not like a book, which must be accessed in a linear fashion, page after page. Instead, it is like having an airplane and an instructor for each student.

The general design used to develop the ECS Trainer is shown in Figure 7. The system is designed to include an instructional environment, a student model, an expert model, and an instructor model. This general architecture is called an Intelligent Tutoring System (ITS). The ITS monitors student performance (through the student model) and compares these actions to the subject matter expert actions (the expert model). Based on the difference between the student and expert actions, the ITS provides remediation and instruction accordingly (the instructor model). When the ITS

technology is combined with the simulation-based instructional environment the power of the instruction is very high. It is similar to providing each learner with a full-time simulator instructor.



**Figure 7.** Intelligent Tutoring System

The remainder of this section will explain each of the components of an ITS and how each is implemented for the ECS Trainer.

### 3.2.1 Instructional Environment

The instructional environment in which the student operates is an interactive simulation of the ECS. The model of the ECS is based on changes in inputs to the system. For example, as the demand on the cooling pack changes, the change is propagated from component to component and reflected on the equipment displays. This simulation gives the student the look and feel of the real maintenance environment.

The student is allowed to manipulate the displays, test equipment, and control panels in any order. When parts are replaced, the simulation will change accordingly. As in the "real world", the student must observe current indications to determine whether the malfunction has been eliminated.



### 3.2.2 Student Model

The student model tracks student performance (ref. 7) within the instructional environment. The student model logs information about what the student has seen and what actions the student has taken for each malfunction. Also, the student model records the amount of time and/or dollars expended to correct a malfunction and the number of errors made.

Some of the types of information stored in the student model are the types of tests performed, which parts were inspected, which parts were replaced and which errors were made. This information is then passed along to the expert and instructor models.

### 3.2.3 Expert Model

As the student troubleshoots a malfunction, the student's actions are compared against those of an ECS expert. This expertise is contained in the expert model. The ECS expert model knows the procedural and/or logical steps that are required to troubleshoot the system. This knowledge is represented in the form of component connectivity diagrams, fault trees, and operational procedures.

### 3.2.4 Instructor Model

When the student's actions are compared against the expert's actions, the instructor model gauges the difference between these two set of actions and decides how to act upon the difference. For example, the student may respond to a Warning light on the ECS Overhead Panel incorrectly. The instructor model must decide whether to let the student make this error and continue, or let the student know of the error immediately. The way that the instructor model responds is usually dependent on the severity and consequences of the inappropriate action.

The instructor model also is responsible for problem selection. The instructor model will analyze the historical information about the problems already completed by the student. The instructor model looks for deficiencies on previous malfunctions to determine which malfunction to present to the student next.

## 4.0 Future Considerations of Advanced Technology for Aviation Maintenance

### 4.1 AMT Schoolhouse Training and Testing

Advanced technology is becoming more widely available for aviation maintenance training. Planned changes to Federal Aviation Regulation (FAR) Part 147, Aviation Maintenance Technician Schools, will permit the use of educational technology in the classroom and laboratory. It is reasonable to expect that computer simulations may supplant requirements to operate certain turbine engines. Such a substitution is critical for schools located in the heart of a city. The use of computer simulations will also be critical in controlling equipment costs for all FAR Part 147 schools.

Technical instructors always customize their materials to meet the needs of their students. Advanced Technology training must be capable of such modification if it is to gain wide acceptance. Most AMT instructors are not computer programmers or computer-based training developers. Therefore tools, usually called authoring systems, must be designed so that AMT instructors can make changes to the advanced technology software. A U.S. Air Force research program called Microcomputer Intelligence for Technical Training (ref. 7) has shown that such authoring systems are possible.

As AMT certification procedures (FAR Part 65) evolve with the new training technology, it is likely that some practical testing may also use computer simulation. Many portions of the General, Airframe, and Powerplant practical exam present the examinee with system troubleshooting problems. Computer simulation is an ideal method to evaluate an individual's system understanding through troubleshooting.

#### 4.2 Training and Job Aiding

This paper has focused on the application of advanced technology to maintenance training. The same technologies can be applied to intelligent job aids. The development of the expert system knowledgebase, for training or for aiding, is labor intensive; therefore, it is very expensive. By designing systems that can be used in training environments and on the job, the cost can be easier to justify. When students became familiar with the system for training, they are more likely to use the system when in the job environment.

#### 4.3 More Technology - Not the Only Answer

This paper has discussed advanced technology from the standpoint of what it can do to enhance human performance in maintenance and inspection. The focus has been on function more than on form. Hardware systems such as digital video interactive, CD-ROM storage, notebook color computers, voice input/output devices, and other such devices have not been discussed. Hardware technology is very important. However, attention to software design and interface design, combined with a knowledge of the capabilities of the human maintainer, are the key ingredients to effective advanced technology applications. Human factors research and development will ensure important attention to the human as advanced technology is introduced to the maintenance system.

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## REVISION OF CERTIFICATION STANDARDS FOR

### AVIATION MAINTENANCE PERSONNEL

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#### SUMMARY

Part 65, Subparts D and E, of the Federal Aviation Regulations (FAR) identify the certification requirements for aviation mechanics and aviation repairmen. The training, experience, privileges, ratings, recordkeeping, and currency requirements for aviation maintenance personnel are also addressed by these parts of the FAR. The recent emergence of the aging fleet problem and the introduction of new technologies, aircraft, engines, and aeronautical products has caused certain portions of these rules to become obsolete. Further, international political arrangements, such as bilateral airworthiness and maintenance agreements, International Civil Aviation Organization (ICAO) standards, certain international agreements for maintenance personnel training, and mechanic certificate reciprocity, have all impacted on the current regulatory policy.

#### INTRODUCTION

Part 65, the primary body of regulations directly relating to the certification of aviation maintenance technicians and repairmen, has not been revised for more than 23 years, and certain, related regulations specifying requirements for aviation maintenance technicians and repairmen have undergone only minor revisions since that time. These minor regulatory changes were accomplished to reflect and integrate more closely with changes in other, related regulations such as Parts 43, 91, 121, 135, 145, and 147 of the FAR. The potential for confusion in the interpretation of Part 65 and other, related regulations has become more evident during the increase in aviation activity driven by deregulation, recent national aviation safety inspection program inspections, and the current FAA regulatory review of Part 147--Aviation Maintenance Technician Schools, (AMTS). In addition, the emerging problems of the aging aircraft fleet have demonstrated a critical need for an extensive evaluation of the current aviation mechanic system of ratings, training, and experience. The ongoing introduction of new technologies and aircraft into the aviation system has highlighted the need for a new evaluation of the aviation mechanic certification process and the possibility of incorporating a revised system of requirements and privileges.

Under the existing rule, aviation mechanic applicants can use military experience to qualify for aviation mechanic certificates, but as military training has been becoming more specialized, fewer military applicants are qualifying for the broader privileges of the FAA aviation mechanic certificate, leaving an experience gap for applicants from the armed forces. Moreover, as the result of international agreements, aircraft maintenance has become a global activity, requiring reevaluation of the current system. The following discussion provides examples of problems considered in the previous section:

1. Inconsistent interpretations: In one FAA region, mechanics performing maintenance on certificated public aircraft maintained in accordance with the FAR were denied eligibility for an inspection authorization (IA). The Aircraft Maintenance Division, AFS-300, advised the region that this type of aircraft maintenance experience did, in fact, meet the requirements of this rule for IA eligibility. Had this issue not been resolved in a timely manner, a worldwide aircraft maintenance program for the United States Air Force would have been seriously disrupted.

2. Aging aircraft fleet problems: During the investigation of several recent aircraft accidents and incidents resulting from structural failure, it became evident that more training and possibly a new certification procedure was required to train and regulate persons engaged in nondestructive aircraft structural inspection. Further, since no single nationally accepted certification standard for this type of discipline now exists, there is no single standard for the FAA to use in evaluating the certification or training of these persons.

3. New technology introduction: During recent aviation industry/FAA maintenance panel reviews, it became evident from the discussions that few maintenance personnel have the technical skills or the industry training required to maintain and inspect composite aircraft structures properly. The same servicing difficulties, according to the aviation industry, will also apply to certain avionics, electromechanical systems, computer-based built-in-test equipment, and many solid-state electronic aircraft systems. No specific FAA certification exists for mechanics engaged in maintaining these complex systems.

4. Equipment type, size, and complexity: Many airline and complex aircraft operators have expressed an interest in FAA certification of mechanics for specific models or types of aircraft, particularly for large and/or complex types such as the Boeing 747 or the Lockheed L1011. Many of the inspection criteria necessary to perform maintenance on these type of aircraft are, in fact, specifically related to the particular aircraft involved. Helicopter operators have also expressed concerns that many of



the maintenance operations that are specific to helicopters are of such a complex nature that a completely different FAA rating is required for helicopter maintenance personnel.

5. International/bilateral agreements: Both the United States/Canadian bilateral maintenance agreement and ICAO will be affected by international events, including the sweeping airworthiness rule changes currently under way in Canada and the recommendations for changing ICAO aircraft mechanic standards as proposed by a recent ICAO Aircraft Maintenance Engineer Licensing Panel. Moreover, the increasing international character of aircraft leasing, parts exchanging, and FAA-certificated foreign repair facilities will and are impacting the current status of FAA-certificated aviation mechanics and repairmen.

The FAA also participated in the recent ICAO Aircraft Maintenance Engineer Licensing Panel. The panel discussed the member state mechanic licensing differences and made recommendations for changes in ICAO certification of maintenance personnel worldwide. The major ICAO issues impacting the certification of FAA maintenance personnel involve the degree of specialization of FAA aviation mechanics and the standards for personnel approving aircraft and components for return to service. In some cases, the licensing differences between ICAO standards and FAA-certificated maintenance personnel could affect the acceptance of American aviation products abroad.

The FAA conducted a series of FAA/industry listening sessions for Part 147 AMTS, the Air Transport Association of America, airline representatives, repair stations, mechanic organizations, and FAA personnel. These sessions were primarily concerned with the regulatory review of Part 147 leading to a notice of proposed rulemaking (NPRM), but the current and future status of Parts 43, 65, and 145 of the FAR were examined as well. The listening sessions incorporated an FAA survey that indicated that there was a strong need to examine the certification standards and privileges granted to aviation mechanics under Part 65 since a number of issues related to Part 147 also impacted Part 65.

AFS-300 recently completed the draft NPRM for Part 147 AMTS. The NPRM addresses the AMTS' administrative and curriculum requirements for the training of aviation mechanics. However, while the NPRM is expected to produce a better trained aviation maintenance technician, there has been no change in the current system of ratings, so the graduate aviation mechanic will still be limited to the current airframe and powerplant ratings specified in the existing Part 65.

On a related issue, in order to clarify and broaden the Part 65 requirements for an IA, AFS-300 is developing a notice to explicitly provide that a person performing maintenance on public aircraft in accordance with the FAR will meet the experience requirements for an IA.

## RESULTS AND DISCUSSION

The development of background information to conduct the regulatory review of Part 65 will require a comprehensive job task analysis (JTA) of what an aircraft maintenance technician does, how often a particular task is done, how it is accomplished, and what types of knowledge, skills, and abilities (KSA) are required to accomplish the task. Further, the JTA will determine how many additional new task elements a mechanic must be able to accomplish since the last maintenance technician JTA's were developed over 2 decades ago. The original JTA study, called the Allen Study (after the original investigator, Dr. David Allen), determined that there are over 125 separate task elements a technician was required to perform in 1968. Since that time, obviously, many new tasks have been required of aircraft maintenance technicians. Many of the new task requirements are driven by the newly emerging technologies of digital electronics, fly by wire, composite structures, fan engines, and so on.

The well publicized problems of the aging fleet have also served to point up some gaps in the maintenance sector. Nondestructive inspection and corrosion control programs are just two of a number of new or rapidly changing maintenance specialties that are required to maintain the aging fleet. Many other issues that were never part of the original regulation may impact the forthcoming rule evaluation. Such parameters as proper work station lighting, temperature, maintenance technician color vision, fatigue and duty times may all impact the development of the JTA.

Some of the JTA issues that will be considered include the aging aircraft fleet problems. During the investigation of several recent aircraft accidents and incidents resulting from structural failure, it became evident that more training and possibly a new certification procedure was required to train and/or regulate persons engaged in nondestructive aircraft structural inspection. Further, since no nationally accepted certification standard for this type of discipline now exists, the FAA cannot properly evaluate the certification or training of these persons.

## CONCLUDING REMARKS

Based on the need to appraise these factors, the FAA is of the opinion that the existing Part 65 and some portions of its companion regulations are in need of a complete regulatory evaluation. It is important to note, however, that this regulation is quite global in scope and depending on the extent of any modifications that might be proposed, some changes could be required in a number of other regulations. In order

to ensure that any regulatory project would encompass enough material to provide a thorough review, our proposed regulatory evaluation will include, but not be limited to:

- A. An analytical evaluation of the aircraft mechanic's occupation, to include KSA's.
- B. Training requirements.
- C. Certification standards.
- D. Rating system.
- E. Currency requirements.
- F. Limitations.
- G. Experience requirements.
- H. Inspection authorization requirements and limitations.
- I. AMTS integration.
- J. Maintenance standards.
- K. Impact on related FAR sections.
- L. Impact on bilateral international agreements and ICAO standards.



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## The FAA Aging Airplane Program Plan for Transport Aircraft

Transport Program Manager: Dayton Curtis, FAA Northwest Mountain Region, Transport Airplane Directorate  
Aging Aircraft Program Manager: Jess Lewis, FAA Flight Standards Service, Aircraft Maintenance Division

### INTRODUCTION

The FAA Aging Airplane Program is focused on five program areas: maintenance, transport airplanes, commuter airplanes, airplane engines, and research. These programs are complimentary and concurrent, and have been in effect since 1988. They are addressing the Aging Airplane challenge through different methods, including policies, research, procedures, and hardware development. Each program is carefully monitored and progress tracked to ensure the needs of the FAA, industry, and flying public are being met.

### TRANSPORT AIRPLANE PROGRAM

Dramatic improvements in airliner performance, capacity, and safety have led to a significant increase of passenger travel. As early as 1968, the FAA recognized that large passenger transport airplanes were remaining in service longer than had been originally anticipated, and it organized the first conference on aging aircraft. In 1978, new procedures for maintaining the safety of airplane structures were adopted for new airplanes, and by 1981 the FAA had published Advisory Circular (AC) 91-56, relating these new procedures to maintaining the structural integrity of older airplanes. Airplane manufacturers responded to this AC by developing Supplemental Structural Inspection Documents (SSIDs) for their older model airplanes. By 1984, compliance with the first SSIDs was made mandatory by Airworthiness Directive.

The obvious failure of this system was the Aloha Airlines accident. This accident was an indictment of current industry and FAA practices devoted to preventing an accident of this type. Such an indictment dictated that the FAA and the industry it regulates reexamine these practices. The system in place in 1988 had provided millions of passenger miles of safe air transportation past original manufacturers' expected operational life criteria. This system is still providing safe air travel daily. Further examination focuses on improving this system.

### OBJECTIVES

The objective of the Aging Airplane Transport Program is to assure the continued airworthiness of large transport airplanes as long as they remain in commercial service. To accomplish this, the Transport Airplane Directorate will take engineering actions in each of the following areas:

- The susceptibility of the Boeing cold bonded lap joints to corrosion and multiple site damage fatigue will be corrected.
- The adverse human factors resulting from heavy dependence on intensive inspections to maintain safety in aging airplanes will be corrected.
- The lack of standards for the control and prevention of corrosion in airplane structure will be corrected.
- The inability of the FAA's earlier aging airplane action (i.e., the current Supplemental Inspection Program ADs) to prevent the Aloha Airlines accident will be studied and appropriate changes will be made.
- The need for the use of damage tolerance principles in the design of repairs for the current generation of airplanes will be satisfied.

- The need for guidance material for operators and inspectors about how to maintain older airplanes will be satisfied.
- The need for first-hand familiarity by FAA certification engineers of the condition of in-service airplanes will be satisfied.
- The lack of a fatigue test basis for some existing transport designs to predict the onset of multiple site damage will be corrected. The lack of a fatigue test based on some existing transport design will be accounted for by a combination of techniques.
- Means will be provided for the FAA to evaluate the effectiveness of the aging airplane initiative so that the program can be altered if necessary.
- The need for better communication within the aviation community on issues related to the maintenance of airplanes will be studied and addressed.
- The need for better engineering support of FAA maintenance inspectors in the evaluation of airplane maintenance programs will be satisfied.
- The susceptibility of aging aircraft structures to multiple site damage or multiple element damage will be assessed for the eleven major aging transport models. Structures determined to be susceptible will be modified or monitored by inspection to preclude failure.

#### APPROACH

The FAA will work with airplane operators, manufacturers, other US federal agencies, and non-US airworthiness authorities to develop comprehensive programs to deal with the aging airplane issues raised by the Aloha accident. Coordination and identification of expectations and concerns is accomplished via the Aircraft Airworthiness Working Group, which has been established under the auspices of the Aviation Regulatory Advisory Committee. This group is composed of representatives from the Federal government, airplane operators, and airplane manufacturers. The manufacturers and operators provide the labor necessary to design corrective actions within this program. The FAA provides the overall management for the program by establishing its goals and objectives, assuring their accomplishment, generating necessary rules, and monitoring program effectiveness. It has also established a Technical Oversight Group for Aging Airplanes (TOGAA). This body is composed of independent technical experts that critique the FAA's program.

#### PROGRAM ELEMENTS

Lap Joints. This effort will implement corrective action for the cold bonded fuselage lap joints of specific Boeing model airplanes.

Schedule:

Issue ADs for Boeing 727/737/747 Airplanes

completed



Modification and Inspection Program. Aging airplanes will be modified to reduce reliance on intensive structural inspection for known fatigue cracking problems. Inspections will be mandated for aging airplane structures susceptible to multiple site damage or multiple element damage in cases where structural modification is determined not appropriate.

Schedule:

Issue Modification ADs for Eleven Major Transport Models	completed
Issue Inspection ADs for Eleven Transport Models as necessary	1/1/93

Corrosion Maintenance. A mandatory corrosion prevention and control program will be added to airplane operators' maintenance programs. This will ensure that corrosion is maintained at acceptable levels.

Schedule:

Support for Boeing Model Corrosion Program	complete
Issue ADs for the Boeing 707,727,737,&747 Models	complete
Issue ADs for the Seven Remaining Major Transport Models	10/1/92
Continued Engineering Support	on-going

Supplemental Inspection Program (SIP) This task will review and update SIPs for the eleven major transport models. Although already mandated by AD, this action will evaluate these models for deficiencies similar to those exposed by the Aloha Airlines accident.

Schedule:

Issue AD Revision for the Boeing Model 707	complete
Issue AD Revision for the Boeing Model 727	complete
Issue AD Revision for the Boeing Model 737	complete
Issue AD Revision for the Boeing Model 747	complete
Issue AD Revision for the Lockheed Model L1011	1/1/94
Issue AD Revisions for the Remaining Six Models	1/1/93

Repairs Assessment. Existing structural repairs will be assessed to assure adequate fatigue resistance and frequency of inspection. Results will be applied to future repairs.

Schedule:

Joint FAA/Industry Repair Survey	5/1/92
Industry Plan of Action Complete	10/1/92
Issue Regulation for Repairs Assessment Program	1/1/95

Research. An extensive research program will be conducted to improve the effectiveness of aircraft maintenance. Specific subject areas are inspections, repairs, structural fatigue prediction, material durability, and human factors. This Program is managed by the FAA Technical Center with support from the Transport Directorate and FAA offices.

Training. FAA engineers and others will be trained in the use and application of damage tolerance principles as they apply to airplane maintenance.

Schedule:

Conduct Periodic Workshops	on-going
FAA Certification Engineer Training	on-going

Airline Visits. Teams of FAA structural engineers will visit airlines to determine the needs of airplanes in service.

Schedule:

Conduct Airline Visits	on-going
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Certification Standards. The present Transport Airline Certification Standards will be updated to establish minimum fatigue testing requirements for new models.

Schedule:

Issue Final Rule for Revision to FAR 25.571	6/1/93
Issue Advisory Circular to Accompany Rule Change	6/1/93

Structural Audit. The FAA will regulate a process that requires structural audits of older model airplanes. This audit will identify any supplemental inspections or modifications needed to extend manufacturers' expected operational life parameters.

Schedule:

Obtain Recommendations from the AATF(ARAC)	10/1/92
Publish NPRM	4/1/93
Issue Final Rulemaking Action	10/1/93
Annual Review Meetings for Foreign Models (3/year)	on-going
Annual Review Meetings for Domestic Models (8/year)	on-going
Publish AD Revisions	on-going

Transport Seminars. The FAA will conduct workshops and seminars to inform the worldwide aviation community of its concerns and actions regarding aging airplanes.

Schedule:

Aging Airplane Seminar for South America	complete
Aging Airplane Seminar for Caribbean/Central America	complete
Aging Airplane Seminar for Jakarta, Indonesia	complete
Aging Airplane Seminar for northern Africa	2/28/93
Aging Airplane Seminar for central/southern Africa	10/31/93

Maintenance Guide. A generic Aging Airplane Maintenance Guide has been developed by the AATF in conjunction with the Maintenance Program of this Plan.

International Conferences. Since 1988, the FAA has actively participated in conferences and meetings concerning aging airplanes. The FAA continues to participate in support conferences.

Schedule:

Attend ICAO Meeting on Continued Airworthiness	complete
International Pacific Air and Space Technology Conference/29th Aircraft Symposium	complete
Sponsor 3rd International Aging Aircraft Conference	complete
AATF Industry Committee Meeting on Wide Spread Fatigue Damage	complete
Support Conferences (5/year)	on-going

Oversight. The Transport Airplane Directorate will continue to provide advice to and liaisons with industry. Continued close coordination of effort is essential to continued success of this program.

Schedule:

Attend AATF Steering Committee Meetings	on-going
Attend Meetings with TOGAA	on-going
Attend Meetings with Manufacturers and Aircraft Certification Offices	on-going

Resources

The FAA has approved eight full time positions to accomplish the Aging Transport Airplane Program. These people will be located throughout the Large Airplane Directorate, and will attend meetings and conferences, approve SIDs, develop regulatory and policy actions, and write ADs that result from the Agency's Aging Transport Airplane Activity.

The airplane manufacturers and operators, under the auspices of the AATF, provide the engineering talent and experience necessary to conduct airplane structure evaluations and to design and implement corrective action programs.



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## AGEING AIRCRAFT RESEARCH IN THE NETHERLANDS

J.B. de Jonge and G. Bartelds  
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### 1. INTRODUCTION

The problems of Ageing Aircraft are worldwide. Hence, actions to overcome or prevent these problems should be taken in international collaboration.

The Federal Aviation Administration FAA and the Netherlands Civil Aviation Department RLD have signed a Memorandum of Cooperation in the area of structural integrity, with specific reference to research on problems in the area of Ageing Aircraft.

This paper gives an overview of the Research currently carried out on this subject in the Netherlands. The work described is largely done at the National Aerospace Laboratory NLR. The major part is done under contracts with RLD and the FAA, as part of forementioned cooperation agreement.

### 2. GENERAL CONSIDERATIONS

Early in 1991, a special Working Group of GARTEUR (Group on Aerospace Research and Technology in Europe) met in the Netherlands to review and discuss the A/A (Ageing Aircraft) problem and potential Research on the subject to be done by the respective research institutes in Europe.

In their overview, the group concluded that the "Multiple Site Damage" problem is the main issue in the Aging Aircraft problem. This MSD-problem is of particular importance for, but not restricted to, lapjoints in pressurized fuselages.

Further the group concluded that distinction should be made between the acute A/A-problems in the current fleet of (aged) aircraft on the one hand and the aspect of preventing A/A-problems in future aircraft on the other.

With regard to the first category, considerable efforts have been made during the past few years. Task Groups, involving both the industry and the operators reviewed the various aircraft types. Supplemental Inspection Programs were specified and "terminating actions" were defined.

Broadly speaking, it appears that the A/A-problem for the existing fleet is under control.

Hence it was concluded that the research effort of the Institutes should concentrate on the second category, that is the prevention of A/A-problems in future aircraft. Potential research areas were considered to be:

- Improved structural design, e.g.:
  - improved structural joints
  - improved (fracture mechanics) analysis tools
  - use of advanced materials, e.g. mixed laminates
- maintenance/inspection techniques and procedures:
  - NDI techniques development
  - automated inspection techniques
  - periodic overloads of pressure cabins as a means to delay crack initiation and crack growth and, thus, to extend inspection intervals
- Increased knowledge of operational loads and aircraft usage.

These considerations have served as a basis for the current research plan on Ageing Aircraft as carried out in the Netherlands.

### 3. A/A-RELATED RESEARCH PROJECTS AT NLR

The current research projects related to A/A at NLR can be divided into two main groups, namely:

- i. Research related to MSD in fuselage lapjoints.
- ii. Research on operational flight loads.

The figures 1 and 2 present the different projects under these headings. The projects will be briefly described in the following sub-chapters.

#### 3.1. Research related to MSD in fuselage joints

##### 3.1.1. Bi-axial tests on realistic fuselage lap joints

A large test program is underway on fuselage lapjoints under bi-axial loading. The majority of joint configurations to be tested will be representative for the Fokker F28 fuselage structure. In addition, USA made specimens will be tested under bi-axial loading, to compare results with those obtained in tests done in the USA under uni-axial loading. Figure 3 summarizes the main test variables. Figure 4 shows the load introduction in the NLR-specimen by means of unidirectional aramid fibre composite slabs that are attached to the specimen by bonding.

The objectives of this rather extensive test programme are:

- to get an improved understanding of the start and development of MSD in realistic riveted joint configurations;
- to determine the effect of bi-axiality;
- to obtain quantitative information on initiation life and crack growth, including the amount of scatter;
- to check the reliability of the available NDI-techniques for MSD detection.

##### 3.1.2. Analytical modelling of MSD

The susceptibility of a lapjoint to MSD depends on the configuration of that joint. A relatively simple "MSDS-parameter" (Multi Site Damage Sensitivity) has been defined, relating the growth rates of larger and smaller cracks. It is a function of:

- rivet pitch, diameter, and type;
- number of rivet rows and sheet thickness;
- friction;
- applied load.

A first check, using recent FAA test data gave promising results. This MSDS-concept will be further developed.

In addition, a recent computer program to predict MSD, prepared for FAA by FractureResearch Inc., will be further developed to include other configurations and stress distributions. Available MSD test data will be used to validate the analytical tools.



### 3.1.3. Statistical data on crack initiation in riveted joints

Scatter is of crucial importance in MSD: If there were little or no scatter in crack initiation life, a uniformly loaded lapjoint showing no cracks at one inspection might show cracks near all rivets at the next inspection!

Unfortunately, systematic data on scatter in riveted joint fatigue life appear scarce. A literature survey of existing data and an analysis of the currently generated test data will be made. If necessary, a test programme on simple riveted joint specimens representative of current fuselage structure will be proposed to generate further statistical information on scatter.

### 3.1.4. Periodic overloads as a means to improve fatigue performance

Periodic high loads are known to cause residual compressive stresses in notches or at crack tips, retarding crack initiation or crack growth.

A study into the feasibility of periodic overloading (say up to 1.3 p once per 10,000 flights), as part of normal maintenance procedure, to improve the fatigue performance of fuselages is foreseen\*.

## 3.2. Research on operational flight loads

### 3.2.1. Participation in the FAA flight loads program

As shown earlier in this Conference, FAA has initiated a very ambitious program on aircraft flight loads. The NLR participates in this program through two different tasks, described as:

1. Acquisition, review and publication of existing European sources of flight load data.
2. Rendering advice on load measurement projects, both for transport and commuter aircraft, with regard to the instrumentation, data analysis and data interpretation.

The European flight data acquired so far include:

- Fokker F27/F28 fatigue meter data.  
For many years, at least two aircraft of each operator were equipped with a counting accelerometer. The acquired data, covering millions of flights, provide valuable information about the variation in load experience between different operators in the short haul- and commuter environment. Fig. 5 gives an illustration of the type of information obtained. Note that for larger  $\Delta n$  values the load factor experience per flight may differ by a factor 10 in severity between different operators.
- NLR B747 ACMS load data.  
NLR has built up a very detailed load data base, covering about 100,000 flight hours from B747 ACMS recordings.
- ONERA "rare event" ACMS data base.  
The ONERA (France) has built up a large data base, covering more than one million flights and a variety of aircraft types on load occurrences in which the

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\* It must be stressed that the present concept should not be confused with so-called proof testing. In our opinion previous studies have shown that proof testing as a cheap alternative for structural inspection is not a viable concept.

incremental load factor exceeded 0.5 g.

- RAE fatigue meter database.

Some decades ago, the RAE (UK) set up a load data base from counting accelerometer data in combination with recorded speed and altitude. This data base, covering 30, 000 flights, includes a large amount of information on load experience at relatively low cruise altitudes.

The acquired load data will be re-analysed using a format that is compatible with existing US data sources (e.g. NACA VGH data) according to the so-called Pratt formula. A PSD-based approach recently proposed by Dr. Houbolt (formerly at NASA) will be used also. This method is presented in AGARDograph 317 "Manual on the Flight of Flexible Aircraft in Turbulence".

It will be clear that the analysis procedures chosen for the "old" data should be compatible with the analysis format selected for the "new" data that will be obtained in the planned FAA-data acquisition program.

### 3.2.2. FOKKER F100 tail load measurements

Flight load statistics available are largely restricted to c.g. acceleration data. Little information on the service load experience of empennage structures is available. Some years ago, the NLR started a project, sponsored by the Netherlands Agency for Aerospace Programs, to measure tail loads on a commercially operated Fokker F100. The instrumentation used consists of a simple 4-channel recorder with solid state memory recording 3 strain gage signals plus lateral acceleration at the tail. Additional information, like c.g. acceleration, speed, altitude, flap position, etc., are derived from the on-board ACMS recorder. Figure 6 shows the instrumentation. Histories of some of the recorded parameters during one flight are depicted in Fig.7. The loading of the stabilizer is presented as "Asymmetric Bending Moment" and "Symmetric Bending Moment". The measurements, which were fully supported by the Fokker company and the operator KLM, were interrupted due to selling of the aircraft but they may be resumed in the future.

### 3.2.3. Load monitoring system for Nusantara Aircraft Industry

The NLR supports the Indonesian aircraft manufacturer IPTN in the definition, procurement, installation and operation of a load monitoring system to be installed in 3 (operational) CN235 aircraft.

The objectives of this project are:

- Verification of the CN235 Design Fatigue Load spectra.
- Build-up of statistical data on the gust experience in the tropical environment of the Indonesian Archipelago.

The instrumentation consists of a nine-channel data recorder with solid state memory and on-board data reduction (peak/valley detection etc.). The following quantities are recorded:

- bending moments in wing root, vertical tail and horizontal tail;
- c.g. acceleration;
- cabin pressure;
- speed and altitude.

Additional information about flights will be obtained from written forms.

The gust data analysis procedure is compatible with the one to be applied to the FAA

flight loads data.

As a next step, it is intended to define a simplified version of the recorder system, which may be used as an "operator oriented" fatigue load monitoring system.

#### 4. SUMMARY AND CONCLUSION

The present paper gives a brief review of current research at NLR related to the problems associated with Ageing Aircraft.

This research is primarily directed towards the prevention of A/A problems in future aircraft and is concentrated on two areas:

- prevention of MSD in lap joints;
- improved knowledge on operational load experience.

The major part of this research is carried out in some form of international cooperation.

The Memorandum of Cooperation between FAA and the Netherlands Civil Aviation Department in the area of aviation's safety provides an effective framework for the coordination of research efforts related to A/A problems.

NLR experiences this cooperation with and support from FAA as very stimulating.



Figure 1

## RESEARCH ON MSD

- BI-AXIAL TESTS ON REALISTIC FUSELAGE LAP JOINTS
- ANALYTICAL MODELLING OF MULTIPLE SITE DAMAGE
- STATISTICAL DATA ON CRACK INITIATION IN RIVETED JOINTS
- PERIODIC OVERLOADS AS A MEANS TO IMPROVE FATIGUE PERFORMANCE

Figure 2

## RESEARCH ON FLIGHT LOADS

- PARTICIPATION IN FAA-FLIGHT LOADS PROGRAM
- SERVICE LOAD MEASUREMENTS ON F100 TAIL STRUCTURE
- ASSISTANCE WITH DEVELOPMENT LOAD MONITORING SYSTEM NUSANTARA AIRCRAFT INDUSTRY

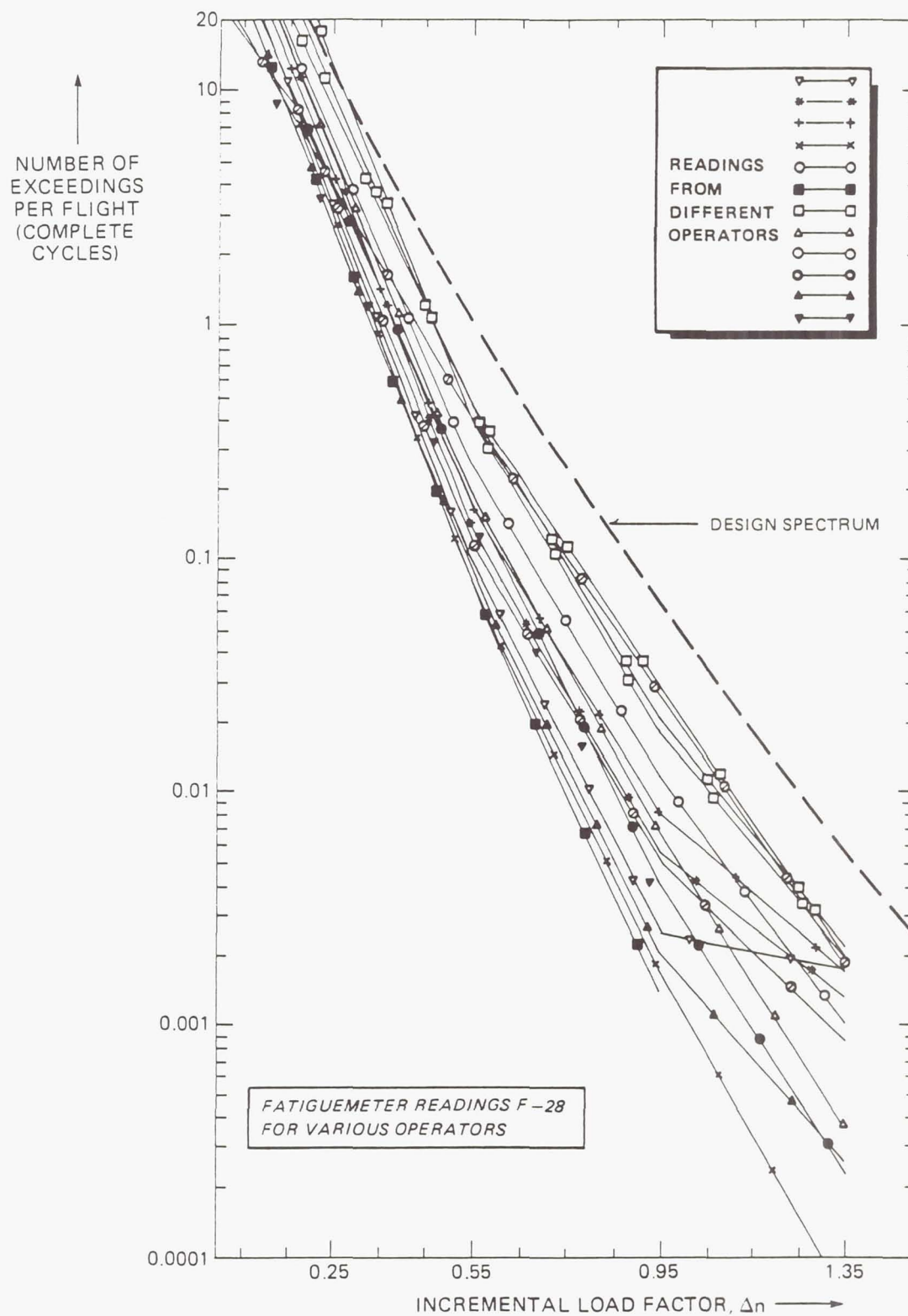
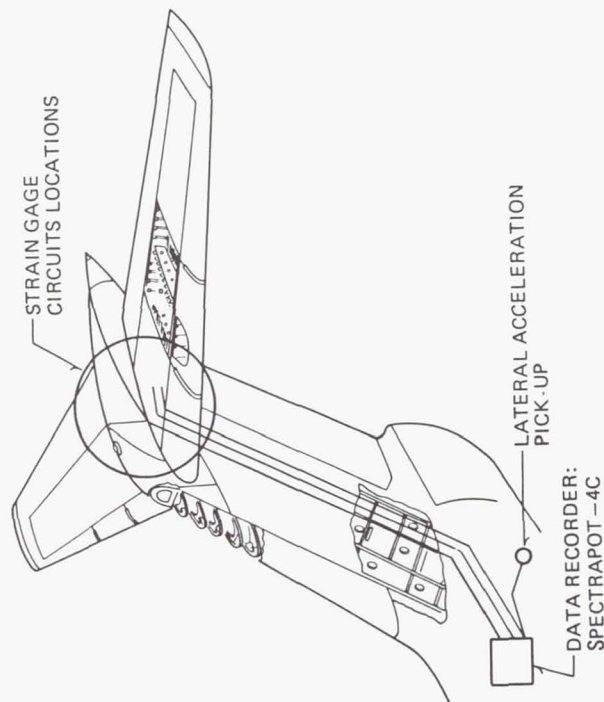


Figure 5 Variation in load factor experience between F-28 operators

Figure 6

# REVIEW OF F-100 SERVICE-TAILLOAD MEASUREMENTS



## MEASUREMENTS WITH SPECTRAPOT RECORDER:

- LATERAL ACCELERATION OF TAIL
- FROM STRAIN GAGE BRIDGES:
  - B.M. OF RIGHT STABILIZER
  - SYMMETRIC COMPONENT STABILIZER B.M.
  - ANTISYMMETRIC COMPONENT STABILIZER B.M.

## ADDITIONAL INFO FROM ACMS SYSTEM:

- ADMINISTRATIVE FLIGHT DATA
- C.G. VERTICAL ACCELERATION
- SPEED
- ALTITUDE
- ELEVATOR POSITION

Figure 7

# F-100 TAIL LOAD MEASUREMENTS EXAMPLE OF RECORDED PARAMETER HISTORIES

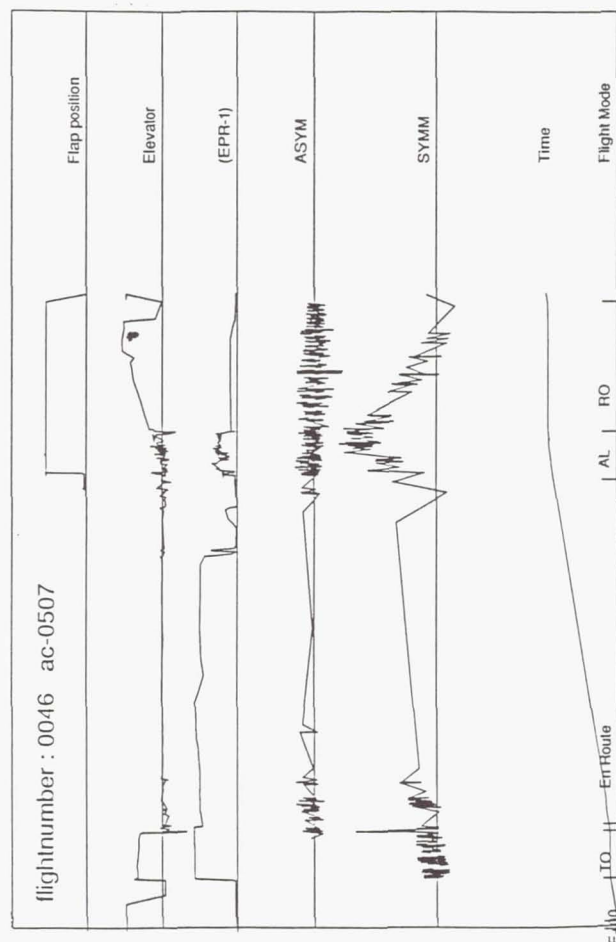




Figure 3

## BI-AXIAL TESTS ON LAP JOINTS

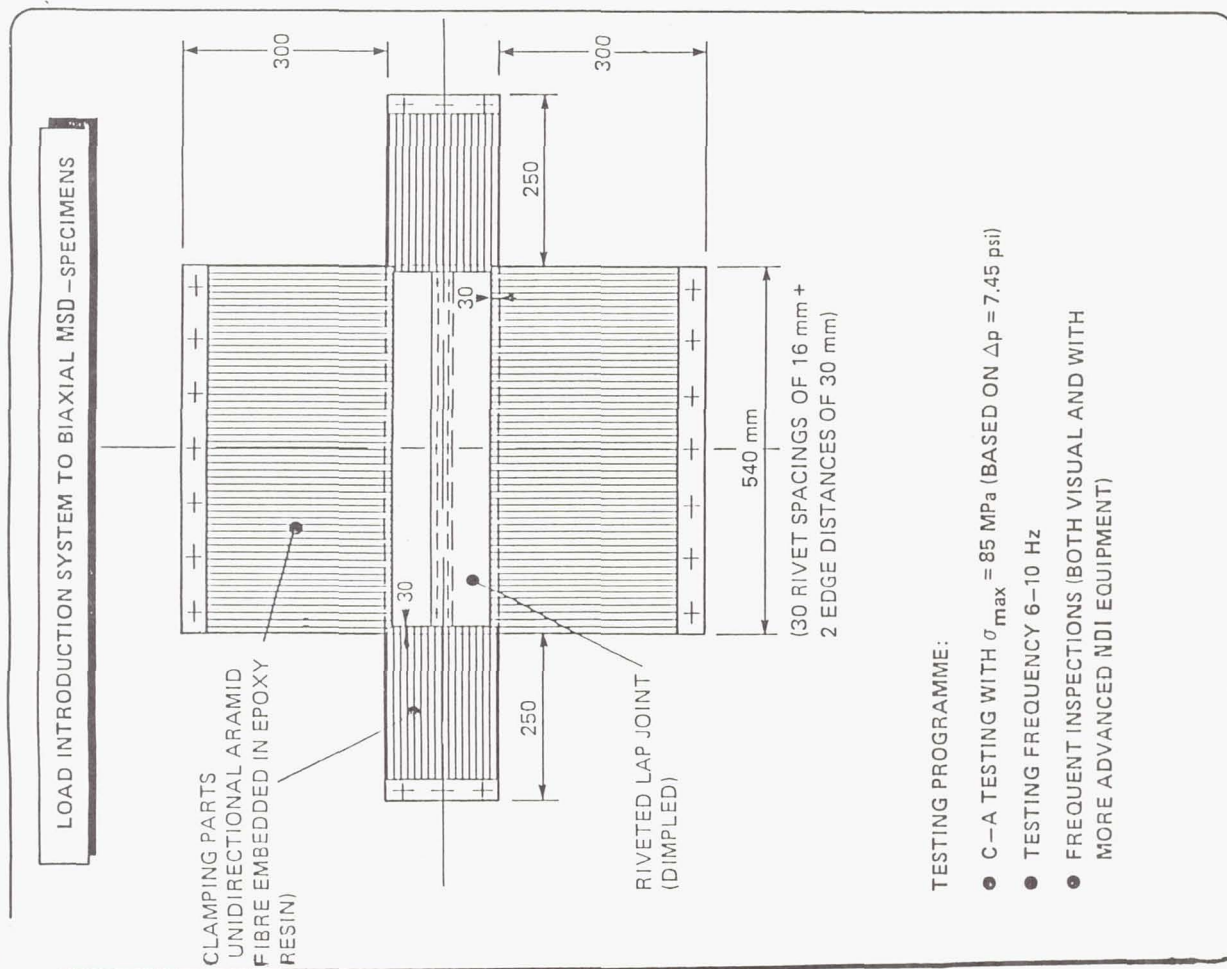
SPECIMEN WIDTH 540 mm (21 inch)

SPECIMEN CONFIGURATIONS:

- 2 AND 3 RIVET ROWS
- DIMPLED AND COUNTERSUNK RIVETS
- BONDED AND UNBONDED JOINTS
- LOCAL DE-BONDS

BOTH BI-AXIAL AND UNI-AXIAL  
C/A TEST LOADING.

Figure 4



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SURVEY OF FRENCH ACTIVITIES  
CONCERNING STRUCTURAL AIRWORTHINESS  
AND AGING AIRCRAFT

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SUMMARY

French activities concerning Structural Airworthiness and Aging Aircraft are presented. A first part is devoted to Basic and Applied Research actions while a second one deals with Full-Scale Testing, Tear-down inspections and continuing Airworthiness approaches.

INTRODUCTION

We present here a survey of actions conducted in France in order to improve the methods and means of accomplishment of the demonstration of Aircraft Structural Airworthiness, with particular emphasis on Aging aspects.

These actions range from basic research up to full-scale testing and tear-down inspections of aged aircraft.

The works cited here are performed by Government Agencies :

- C.E.A.T. : Centre d'Essais Aéronautiques de Toulouse
- O.N.E.R.A. : Office National d'Etudes et de Recherches Aérospatiales.

and Manufacturers :

- Aérospatiale
- Dassault-Aviation
- Messier-Bugatti (Landing gears).

we shall distinguish here :

- o Basic and Applied Research studies
- o Full-scale testing (including Tear-down).



## BASIC AND APPLIED RESEARCH STUDIES

### Part 1 : Metallic Structures.

#### 0 Crack initiation

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In this area it is worth mentioning the development of a methodology for crack initiation computation by Dassault-Aviation. Among the numerous difficulties to be dealt with, special attention was paid to the definition of a parameter able to give a unified presentation of basic test result and consequently leading to master curves like the one presented in figure 1 for the 7010 T73651 aluminum alloy.

Two studies were conducted at CEAT and ONERA concerning the influence of load history on crack initiation. The first one (C.E.A.T.) aimed at the derivation of a method to predict fatigue damage allowing for small cycles (i.e below the endurance limit) which would lead to no-fatigue damage according to the Palmgren-Miner's rule whereas this is not in accordance with certain experimental results. Through two modifications of the Wöhler's curve indicated on figure 2, very good predictions are achieved, compared to experimental results where specimens undergo TWIST, MINI-TWIST and modified MINI-TWIST standard sequences.

Concerning the second one (O.N.E.R.A.) the influence of apparition of an overload was predicted using a damage model developed at O.N.E.R.A which proved worthwhile when compared to experimental results as shown on figure 3.

Material's characterization was conducted at Aérospatiale in particular for Al-Li alloys. Figure 4 shows the main observed tendencies. The lower endurance limit of the Al-Li alloy is attributed to the higher brittleness of its precipitates.

Concerning landing gear structures, the following studies were conducted at Messier-Bugatti :

- Fatigue life prediction under complex load histories
- Investigation of several damage criteria in the case of multiaxial loading
- Evaluation of the influence of factors like design, loading history, processing and assembly modes, etc... on fatigue life of landing gear components.
- Figures 5 and 6 show the influence of surface treatment on fatigue life of an aluminum alloy and a steel.

#### 0 Résidual Strength (R curves)

-----

Careful examination of results from the open literature indicates that Linear Elastic Fracture Mechanics (L.E.F.M.) cannot be an accurate prediction method, particularly in presence of thickness effects. Taking plasticity and energy balance into account results in substantial improvement as demonstrated by the results obtained by Dassault-Aviation and Aérospatiale (see figures 7 and 8).

## O Fretting Fatigue

---

The assessment of fretting-fatigue susceptibility should be taken into account in the early part of an alloy development. The experimental device described on figure 9 is used at C.E.A.T to compare the intrinsic behavior of various alloys under fretting-fatigue conditions and to value surface protections which are intended to delay fretting occurrence. Figure 10 shows the observed behavior for several aluminum alloys and figure 11 points out that 2000 alloys have a better behavior under fretting fatigue than the 7075 one. With the same goals, a study was conducted at Aérospatiale on the 30 NCD 16 steel and aluminum alloys and confirms the conclusions of C.E.A.T. (see figures 12 and 13).

## Fatigue Crack Growth

---

In this field, various studies allowed checking capabilities of several models for predicting crack growth under aeronautical type spectrum loadings. It is worth mentioning here that a comprehensive survey has been achieved through a fruitful GARTEUR cooperation associating N.L.R. (Netherlands), R.A.E (Now D.R.A.) (U.K.), D.L.R (Germany) and O.N.E.R.A (France).

In particular, two databases were created :

- extended F27 spectrum variations (associated with 2024 alloy and thickness effects) were studied.
- FALSTAFF database, with the same investigations as for the F27 one and associated with several materials : 2024, 7075, 7475 and 2214 aluminum alloys, a titanium alloy and a steel.

Figure 14 shows the prediction obtained with the ONERA 's model.

In another cooperation between Boeing Commercial Airplane Division and ONERA, ONERA's model was also tested versus a "wing upper surface" type of spectrum, and results are illustrated in figure 15.

Developments were also conducted at Aerospatiale in order to improve the description of the influence of compressive cycles during spectrum loadings. Special attention was focused, at CEAT, on Al-Li alloys for A330/A340 applications ; it has been noted a good consistency between data on simple specimens and those on specimens representative of a structural feature. A good behavior regarding crack growth and some weakness regarding throughness are confirmed for 2091 sheets (see figures 16 and 17).

## Non Destructive Inspection (N.D.I)

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For metallic materials, works in this domain concern essentially the eddy current technique and significant efforts were undertaken to improve both the methods and the associated hardware. Let us mention the SIAM (System of Inspection-Assisted by Microprocessor) developed by Aerospatiale, for which practical runs on 65,000 rivet samples have demonstrated a capability of up to 1,400 rivets inspected per hour, without any false alarm. AIDA system was developed by DASSAULT AVIATION with the help of TECRAD Society in view of

For failure, special attention was paid at Aérospatiale to the problem of fastener holes in view of prediction of bolted area failure, taking into account parameters like :

- type of loads (tensile, compressive, shear, bending ...)
- geometry (lay-up, diameter, spacing ...)
- environment (temperature and relative humidity).

#### FULL SCALE TESTING AND TEAR DOWN

##### Part 1 : Fatigue and Damage Tolerance Testing

###### o Falcon 900 : (Dassault Aviation and C.E.A.T.)

-----  
Within the type certification framework for this aircraft, a general fatigue and damage tolerance test on its structure has been conducted. 2 lives (40,000 flights) were simulated in a safe-life approach followed by 1 life (20,000 flights) in a damage tolerance approach. It should be noted here that the fatigue load spectrum applied was deduced from in-service load monitoring which improves the significance of such a test.

###### o A 320 : (Aérospatiale and C.E.A.T.)

-----  
1.25 economic life (60,000 flights) was simulated in a safe-life approach, followed by again 1.25 economic life in a damage tolerance philosophy. On the front fuselage part 24 artificial cuts were tested ; the pylon and centre wing box achieved 120,000 flights with 11 and 13 artificial cuts respectively.

###### o ATR 42/72 : (Aérospatiale and C.E.A.T.)

-----  
The design life goal is 70,000 flights and the simulated flight goal is 140,000. For ATR 42 flights 134,000 were achieved on the wing box (15 artificial cuts) and 100,000 on the fuselage and empennage (34 artificial cuts). For ATR72 centre wing box achieved 30,000.

###### o A330/340 : (Aérospatiale and C.E.A.T.)

-----  
The design life goals are respectively 40,000 flights for A330 and 20,000 for A340.

For the forward fuselage, the simulated flight goal is 120,000 and the corresponding test starts at end 91. The wing/centre fuselage will be submitted to mixed typical mission (A330 + A340) a simulated goal of 80,000, test being started at mid-92. The figures for the A340 pylon are 50,000 (flight goal) with test starting at end 91.



investigating cracks in holes ; here again, this system microprocessor-automated which allows, for example, differential and absolute measurement, 400° rotation of probe with reciprocating motion, variable frequency output ... etc and further developments will lead to automatic crack detection and automatic signal compensation to account for edge effect. Finally, CEAT has also conducted in-depth evaluation of the possibilities of eddy current techniques particularly with respect to reliability and statistical signification of this technique.

## Part 2 : Composite Structures.

### 0 Damage Tolerance

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In this domain, the problem of low-energy impact on composite structures has received substantial attention since the damage created in these conditions is often barely visible and can subsequently develop under normal service loadings before being detected.

Several actions were conducted at Aerospatiale and Dassault Aviation:

- identification of main stress effects governing damage
- experimental study of morphology of degradations induced on composites by low speed impact
- study of impact behavior of prestressed panels (simulation of hail for example) either from the prediction side (development of theoretical models) or the experimental one.

It appears clearly, from this analysis, that peeling resistance and shear resistance of the interply play a very important role in the process and should deserve particular attention in future works.

Concerning residual strength, Aerospatiale has carried on instrumented compression tests in order to determine the successive steps leading to the ruin of a pre-impacted plate. Figure 18 illustrates the tests conditions and those of the associated numerical analysis. It was remarked a good agreement between measured and predicted strains and a failure criterion is now under study for complete prediction of residual strength.

Finally, we wish to mention here the probabilistic approach developed by Aerospatiale combining three types of probabilities, namely :

- prob. of encountering an impact of given energy
- prob. of encountering loads between Limit and Ultimate load
- prob. of detecting damage.

In view of defining the residual load a structure must sustain with a risk factor less than  $10^{-9}$  per flight hour.

### 0 Failure and fatigue behavior.

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Concerning fatigue, the main problem to be tackled is the accurate representation of the fatigue spectrum which is required to gain further knowledge in the following areas :

- scatter and standard variation on life
- variation coefficient on fatigue stress
- endurance curve with load-effect, R factor...

## Part 2 : Tear down inspections.

They include the following operations :

- detailed inspection of the structure as it is, implementation of procedures described in the maintenance manual.
- experimentation and development of new NDT procedures
- cutting out of known cracks plus microfratographic examinations
- disassembly of components difficult to inspect and search for hidden damage.

In cooperation between Aerospatiale and C.E.A.T, the following aircraft structures undergo a tear-down inspection after a full scale fatigue test :

### 0 Airbus A310 :

-----

58 critical sites were inspected and about 60 cut-outs were performed (see figures 19 and 20). This structure was subjected to 100,000 simulated flights for a design life goal of 40,000 flights. The zone inspected are :

- the wing centre section
- the centre fuselage

### 0 Concorde SST :

-----

21 sites were inspected and cut-out :

- 9 zones on the fuselage (50 holes cracked in 3 new sites)
- 12 zones on the wing (50 holes cracked in 6 new sites)

The Concorde structure was subjected at about 34,000 simulated flights in the course of fatigue testing, which have so far enabled 6,700 supersonic flights at heavy weight to be authorized in service. Figure 21 gives indications on the location of the inspected sites.

## Part 3 : In service loads

### 0 General :

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During entry into service of an aircraft, maintenance activities are essentially based on analysis. The very first years of operation are also protected with respect to fatigue by the credit opened by the progress of fatigue testing. It is no less true that all the data used for this purpose is based on assumptions as to how airlines operate their aircraft. It is therefore necessary to be able, in one way or another to identify the reality of aircraft operation as accurately as possible, in order to correct the maintenance instructions given to the operators, as necessary.

#### O A300 B2 - B4 campaign

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Among the possibilities that exist in this fields it is interesting to mention the campaign performed by Aerospatiale at Airbus Industries request, on the A 300 B2 - B4 fleet.

The purpose of this campaign was above all to check that the assumptions chosen for establishing typical fatigue missions (which constitute the basis for fatigue and damage tolerance calculations and test)

- take off weight,
  - landing weight,
  - range,
  - payload,
- are not found to be at fault.

About 45,000 flights were analysed, involving five airlines in Europe and the USA, and both A300 B2 and A300 B4 aircraft. The results of this analysis highlighted significant variations with respect to predictions. This

made it necessary to make a comprehensive parametric study of the general loads, based on :

- payload,
- range,
- fuel at landing.

After fatigue calculations on sensitive structural components, this study provided correction laws affecting the inspection intervals given in the maintenance manuals.

#### O A310 campaign

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For the A310, a similar activity was initiated with two European airlines. In addition to checking the typical fatigue mission, the purpose of this campaign was to assess other data such as equivalent gust speeds, moving surface deflection, etc.

The same phenomena as those observed on the A300 exists on the operating weights ; it has already been decided to take corrective actions on thresholds and intervals on the basis of A300 investigations.

### Part 4 : Continuing Airworthiness

#### O General :

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Within the framework of continuing aircraft airworthiness, the development of supplementary structural inspection programmes for the Caravelle SE 210 and the A 300 B2-B4 should be noted.

The aircraft, in particular the Caravelle, were not certificated according to far 25 amendment 45 and must thus, in compliance with amendment 45, form the subject of suitable monitoring whose conditions must be defined while respecting the spirit of damage tolerance.



O Caravelle SE 210 :

-----  
Certification in 1957, entry into service in 1959, the Caravelle in its different versions, has accumulated a total of about 6,300,000 flights. To date there are still 70 aircraft in service in the seven versions sold. The aircraft which have accumulated the greatest number of flights have reached 43,000 flights. The structure in its basic definition was subjected to a fatigue test (104,000 simulated flights) to substantiate a target service life which was 32,000 flights at that time. The experience gained year after year allowed 43,000 flights to be achieved. The supplementary inspection programme was developed and issued, at least in preliminary form, at the end of 1989.

The stages which have been defined are as follows :

- reorganization of maintenance instructions in the form of a maintenance manual,
- analysis of the structure in the spirit of damage tolerance : identification of the Principal Structural Elements,
- analysis of the PSE : identification of the sites forming the subject of special monitoring, and the sites which are not monitored at the present time and are thus candidates for additional inspection tasks.

To date 60 new sites have been selected which requires at least 200 damage tolerance analyses. These analyses are performed on the basis of 4 typical missions defined further to a study of actual fleet operation.

O A300 B2/B4 :

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Certified and put into service in 1974, the A 300 in its different versions, (A 300 B2-B4) has accumulated a total of about 3,600,000 flights.

Although this aircraft had not been certificated to amendment 45, Aerospatiale and its partners decided at that time to develop calculation and test activities with a view to determining propagation duration and the residual strength of the structure tolerance in the presence of damage. Implementation of the damage tolerance concept to set up a supplementary inspection programme has not thus raised any particular difficulties.

The development of the SSIP covers the following operations :

- a) analysis of the fatigue test results and the tear-down operation leading to the fatigue sensitive sites.
- b) analysis of in-service problems which had revealed sensitive sites not identified during tests or damage initiation earlier than the date on which it occurred on the fatigue test.
- c) recording of the sites for which it is necessary to re-evaluate the inspection thresholds and intervals.
- d) taking the analysis of in-service operation statistics into account for correcting the intervals according to mission durations.

For the parts under AS responsibility, 14 service bulletins and 25 new zones to be inspected were thus provided.

When this supplementary programme was implemented, a method for calculating the inspection intervals was developed by Aerospatiale based on :

- the probability of fatigue crack initiation,
- the probability of propagation of this crack,
- and the probability of its being detected during inspection,

This method consists in determining the inspection threshold on interval causing a value of the risk of failure which is not to be exceeded at the end of the aircraft operation.

O A 310 :

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Contrary to the A 300 models, the A 310 models were certificated in compliance with the requirements of amendment 45 and do not thus need to be subjected to an additional inspection programme. Fatigue tests were carried out, up to 100,000 simulated flights with a view to substantiating a target life of 40,000 flights and the tear down operation already mentioned is naturally extended by a revision to the basic programme.

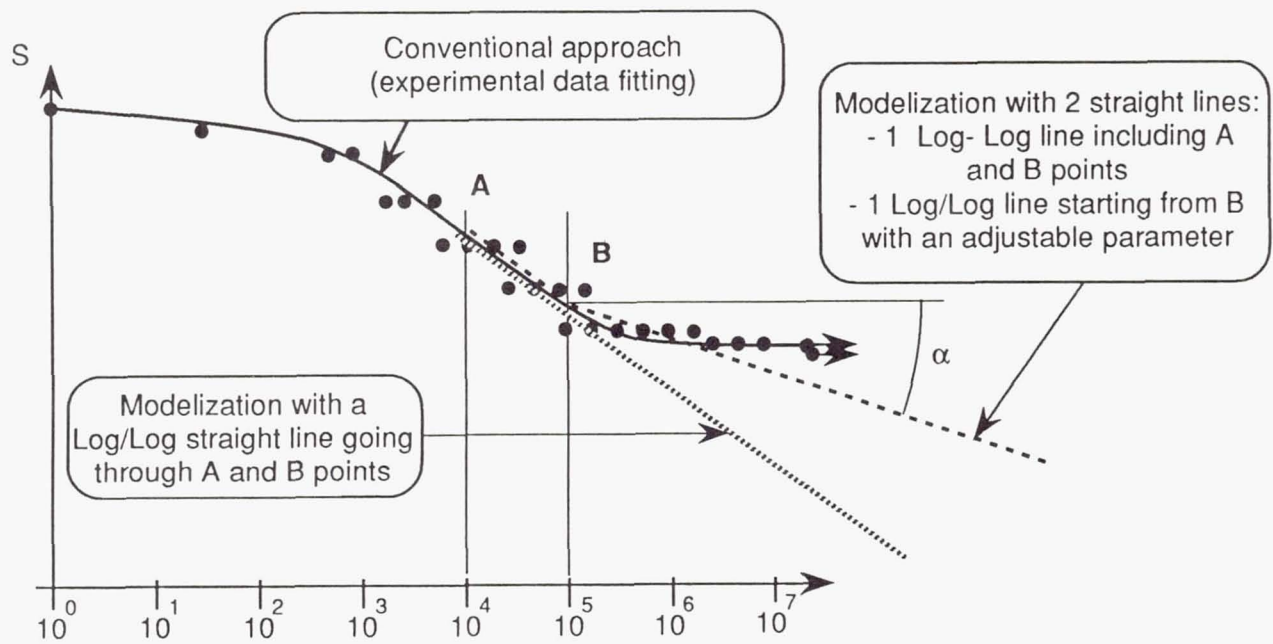
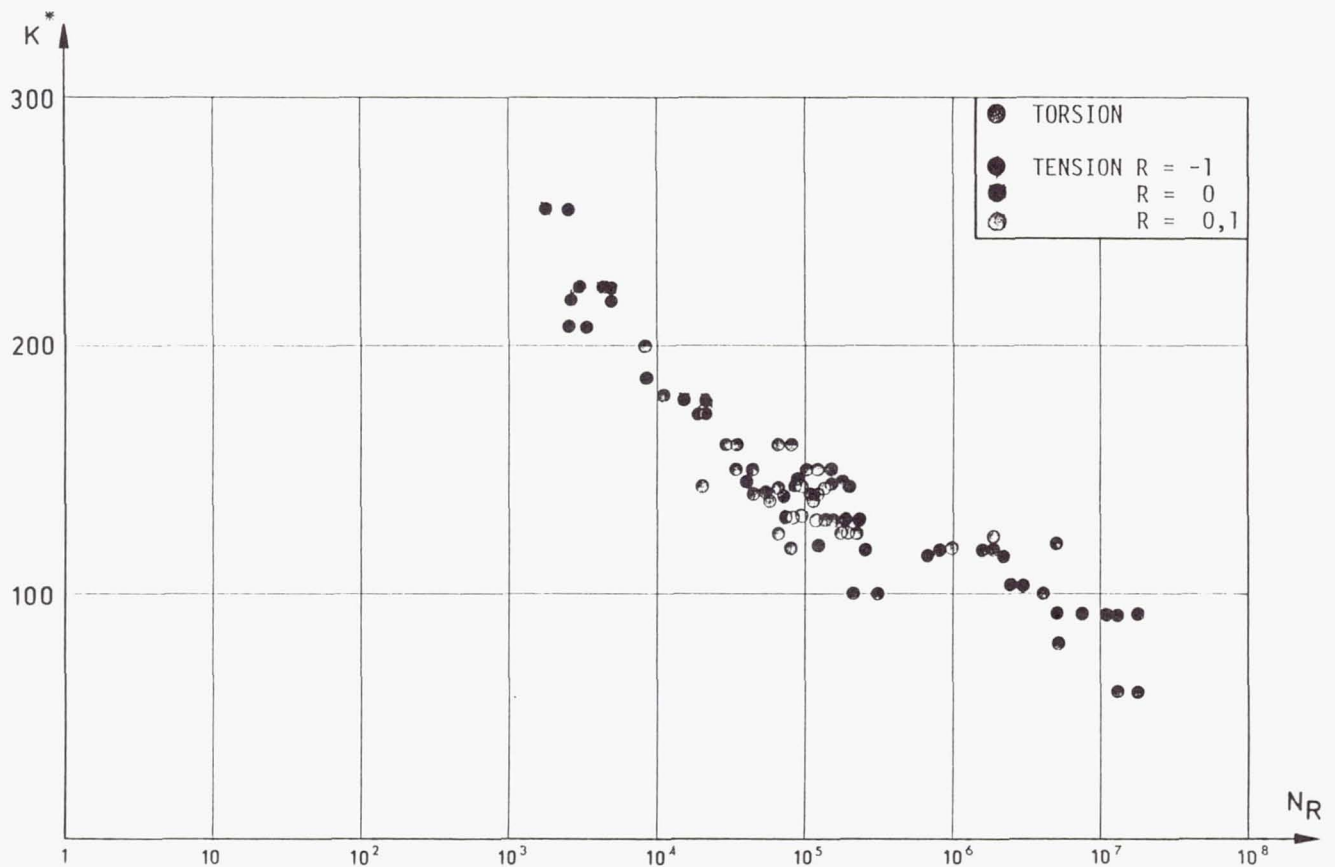
The general outlines of the method adopted are using those developed for the A 300 SSIP and led to some adjustment of the basic inspection program.

#### CONCLUSIONS

The results presented here demonstrate that a very important effort is still maintained for Structural Airworthiness Assessment. Yet, although the safety level with respect to Structural Resistance has dramatically improved over the past, this effort should be maintained and even increased for many reasons :

- Fundamental knowledge, and moreover modelling, of the various types of damage appearing in an aircraft structure is still not sufficiently accurate and this justifies the on-going associated basic research.
- Actual loads are still difficult to predict even for "classical" aircraft and in-service experience still continues to reveal non-predicted sites of damage and poorly predicted initiation and damage growth rates.
- This should continue with the coming in service of modern technology aircraft where the fly-by-wire/Computer assisted commands will generate, at least partly, new loading spectra.
- The emergence of new materials and fabrication processes will induce new types of behavior with specific manifestations of damage.

Finally we wish to stress that, even if national resources should be activated at their maximum, we are convinced that international cooperation is a major key on our way towards progress.





Constant amplitude loading  $\sigma = \pm 230.6$  MPa  
 Overload  $\sigma = \pm 410$  MPa

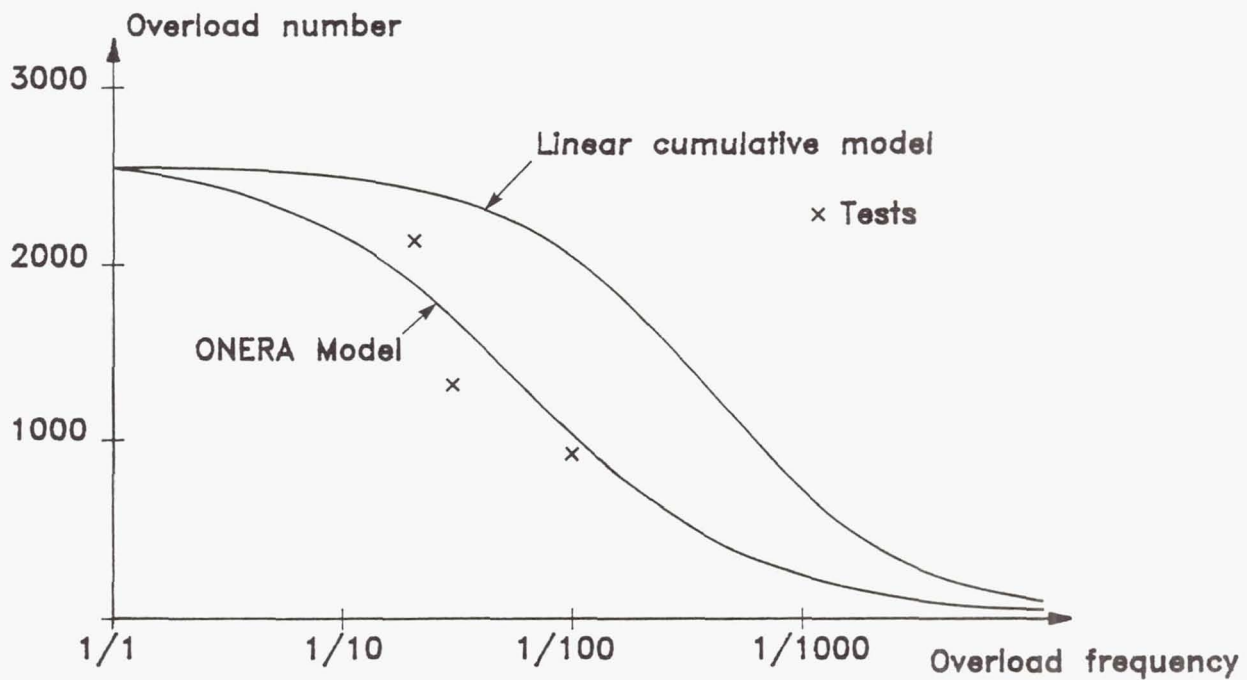


Fig 3 - Influence of overload

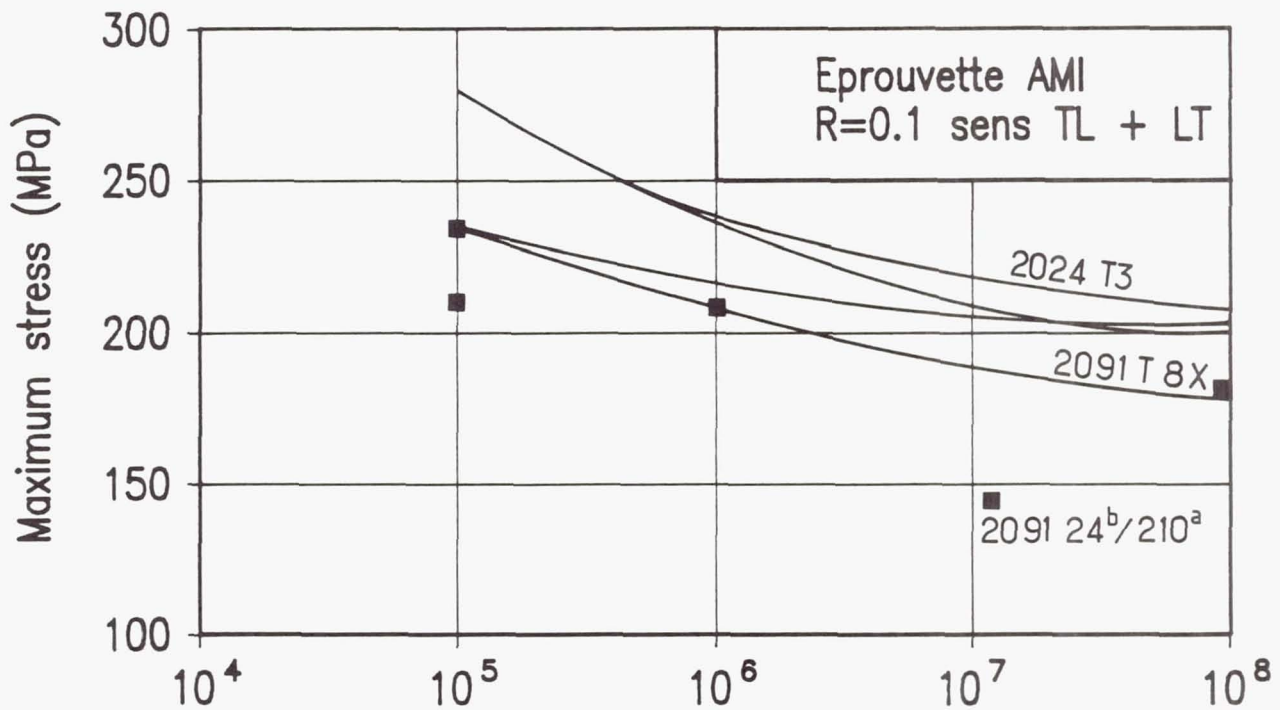


Fig 4 - Fatigue crack initiation in 2024 and 2091 aluminium alloys

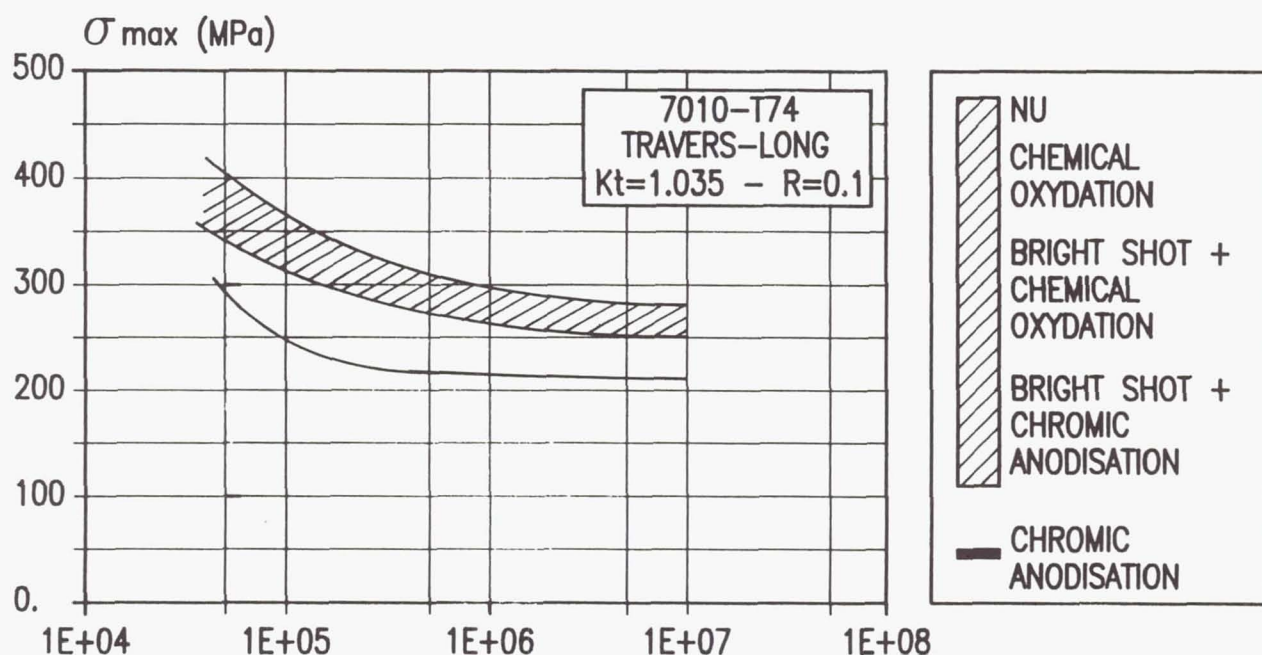


Fig 5 - Fatigue behavior with various surface treatments (aluminium alloys)

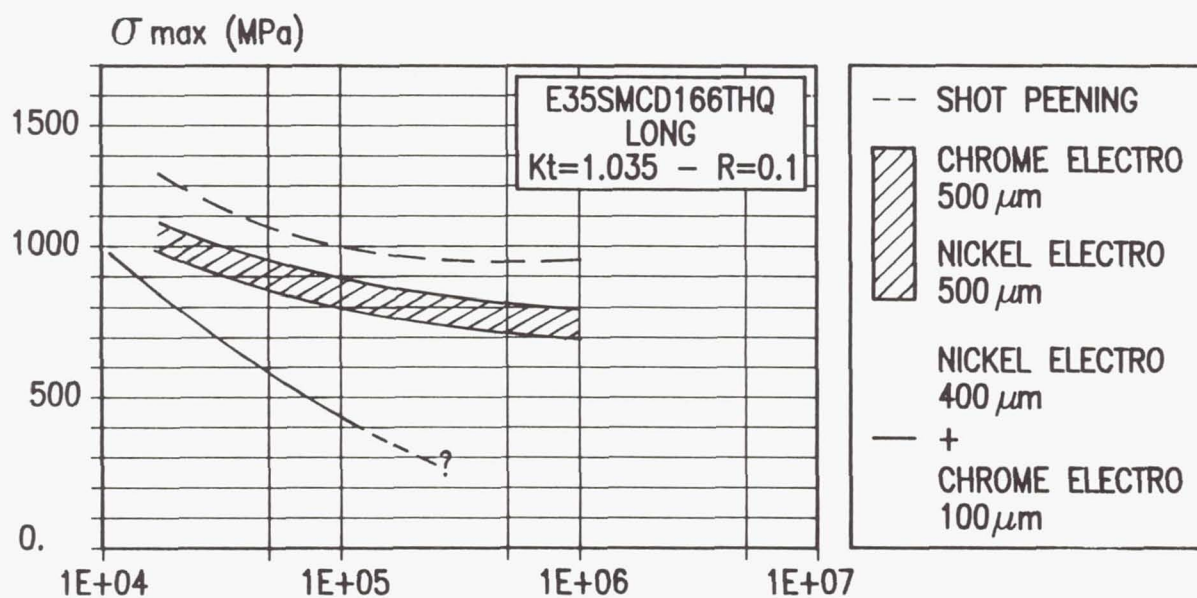


Fig 6 - Fatigue behavior with various surface treatments (high strength steel)

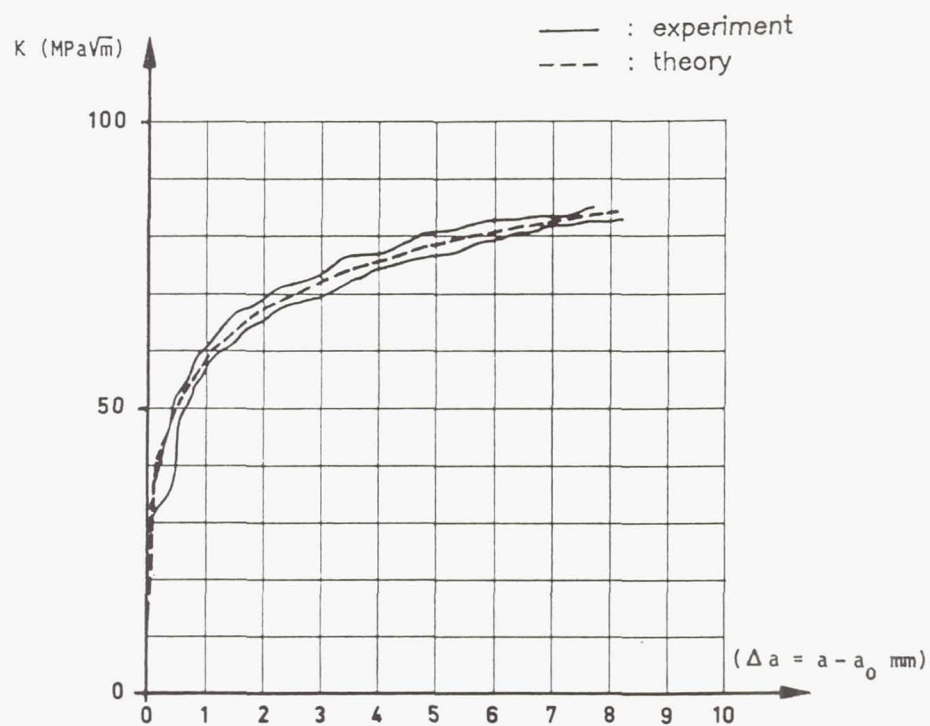


Fig 7 - Fracture toughness : comparison between theory and experiments.



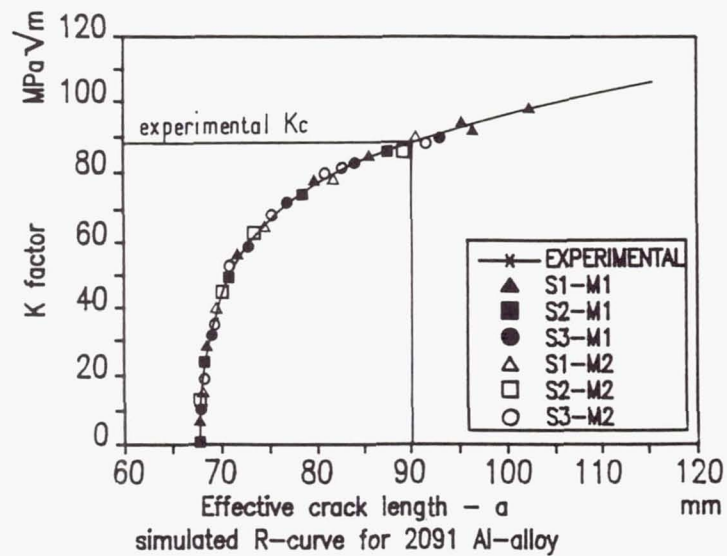
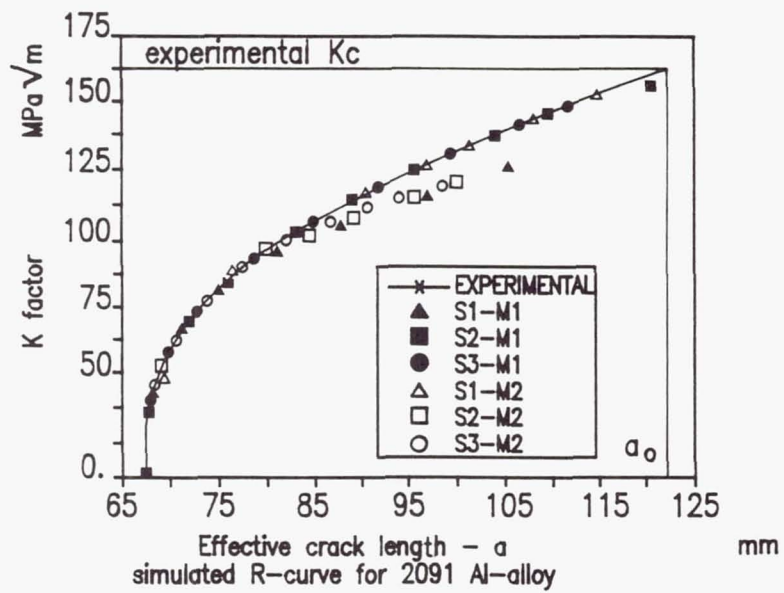
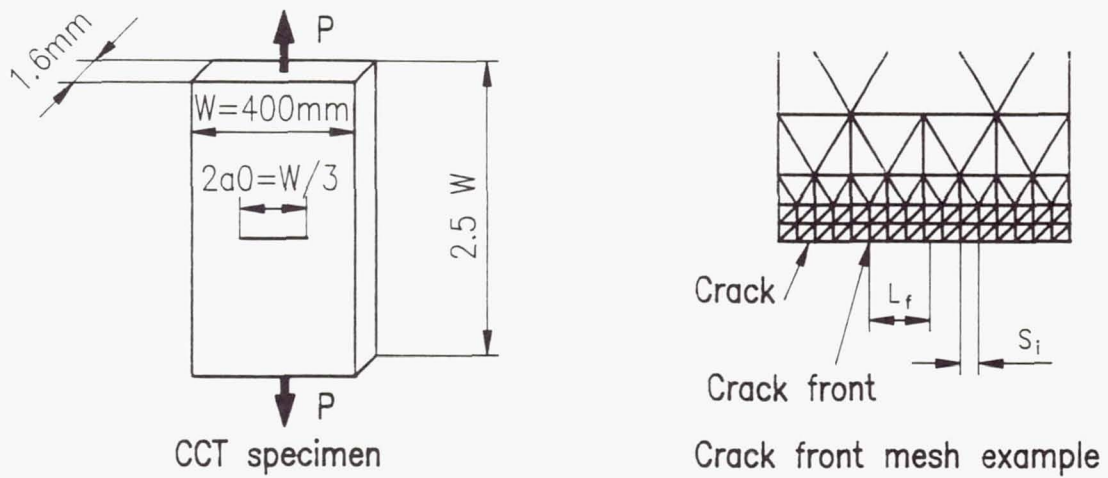


Fig 8 - R curves simulations for 2024 and 2091 aluminium alloys.

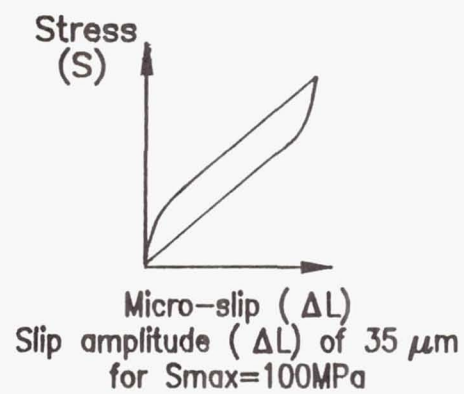
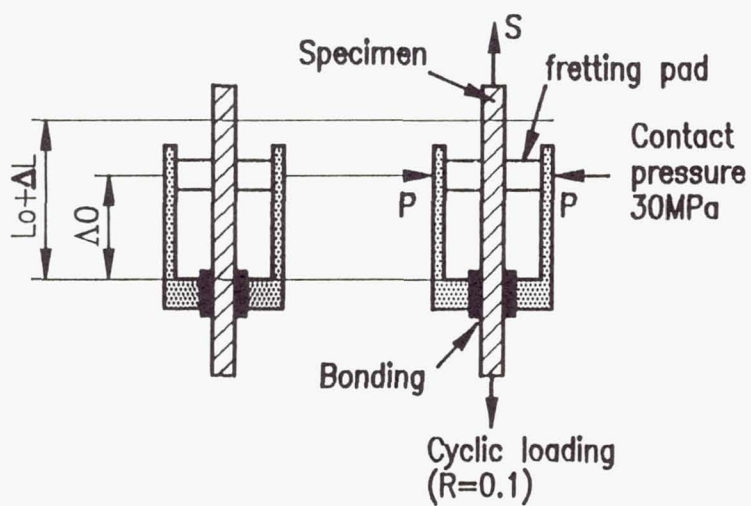
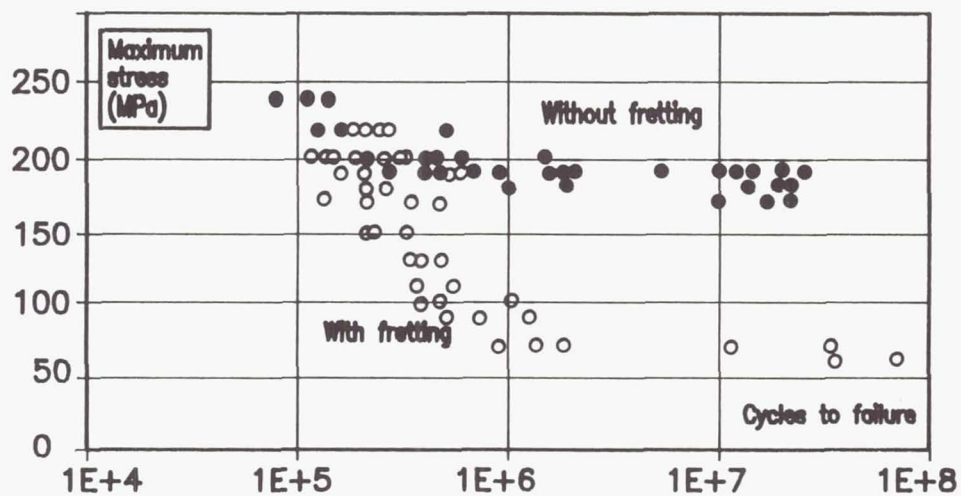
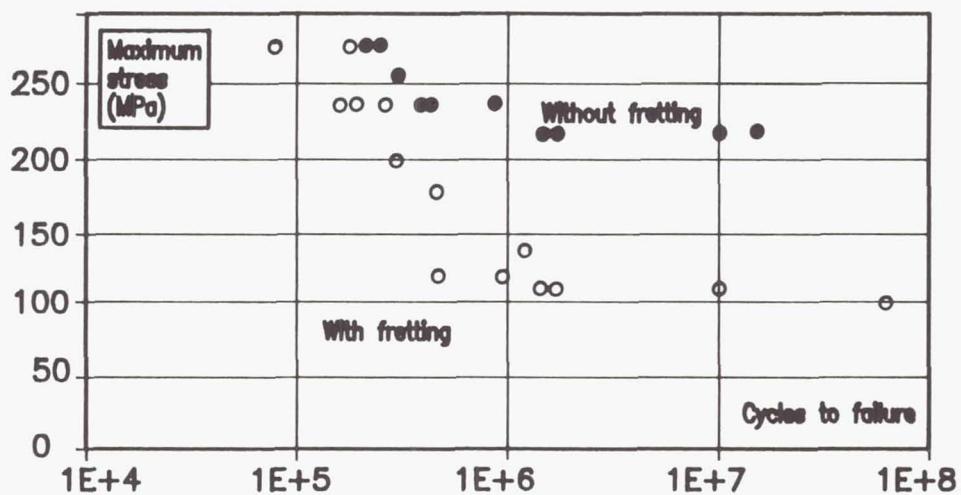


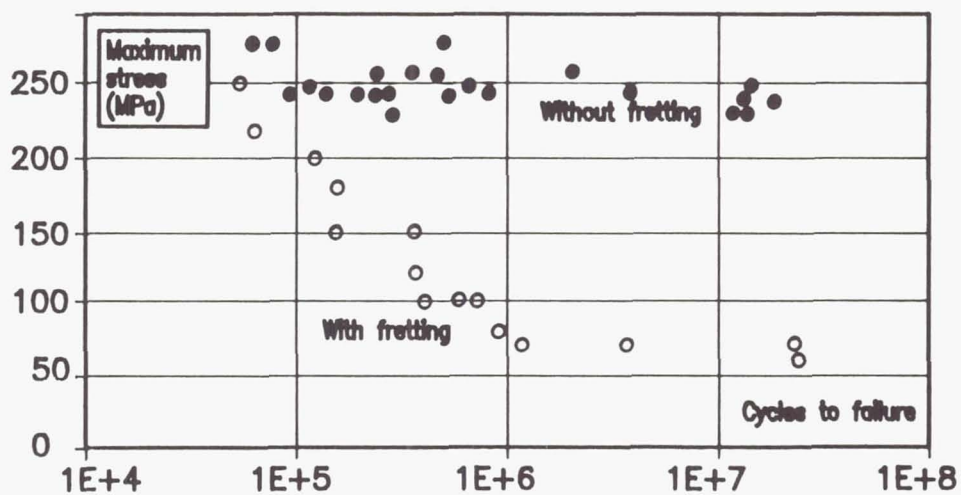
Fig 9 - Fretting experimental device



2024 T351



2091 T8x51



7075 T7351

Fig 10 - Behaviour of various aluminium alloys under fretting fatigue



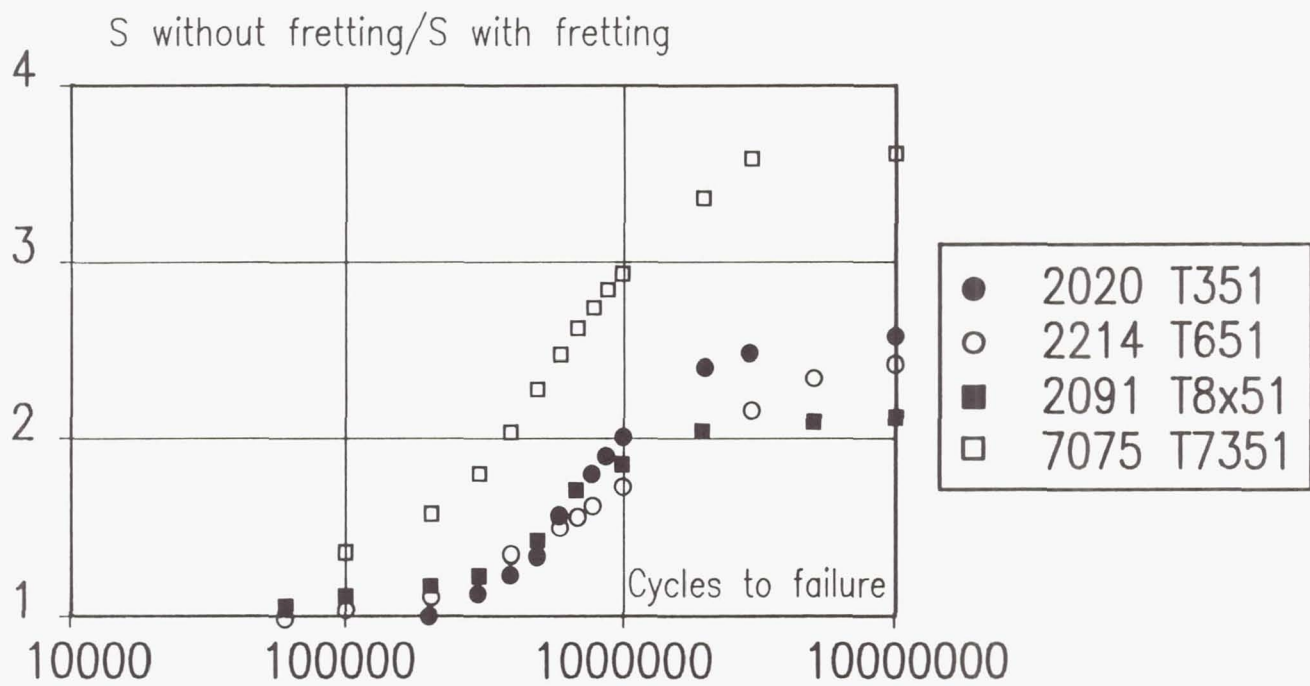


Fig 11 - Influence of fretting on fatigue resistance



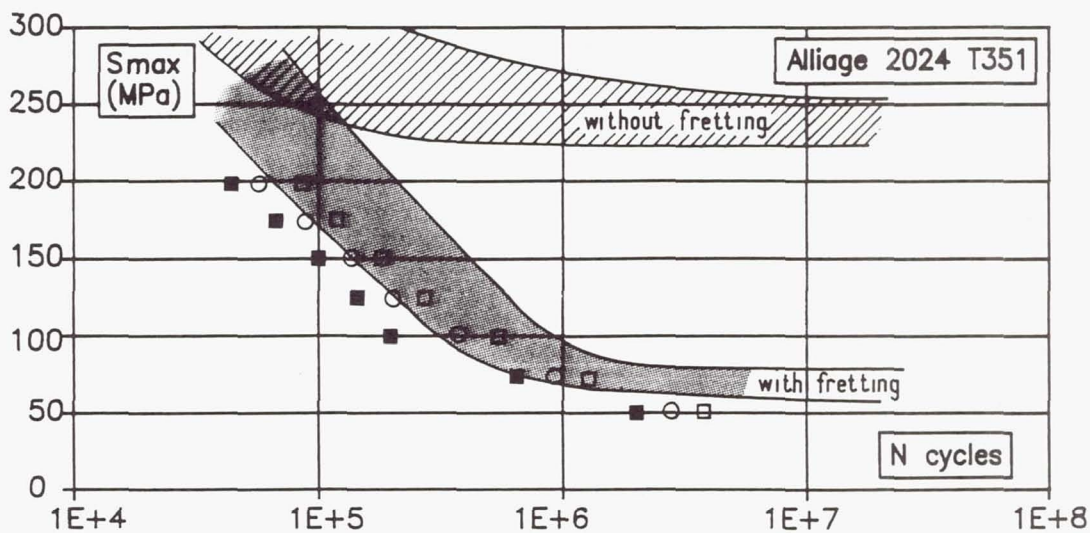
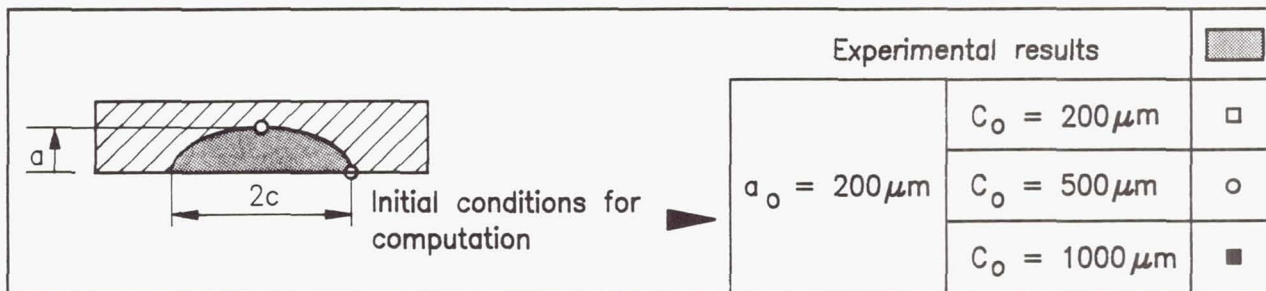
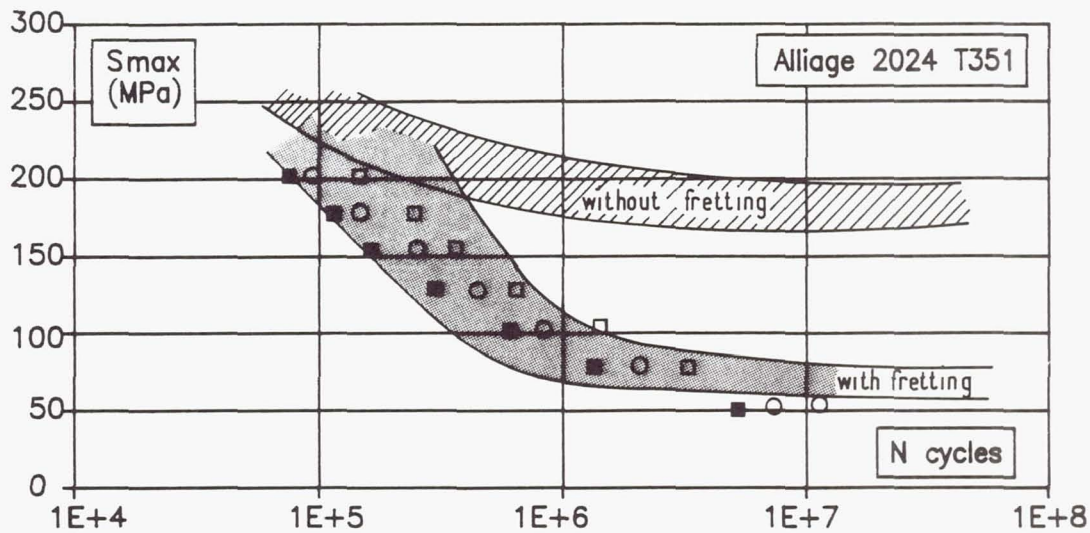


Fig 13 - Crack growth predictions in presence of fretting fatigue



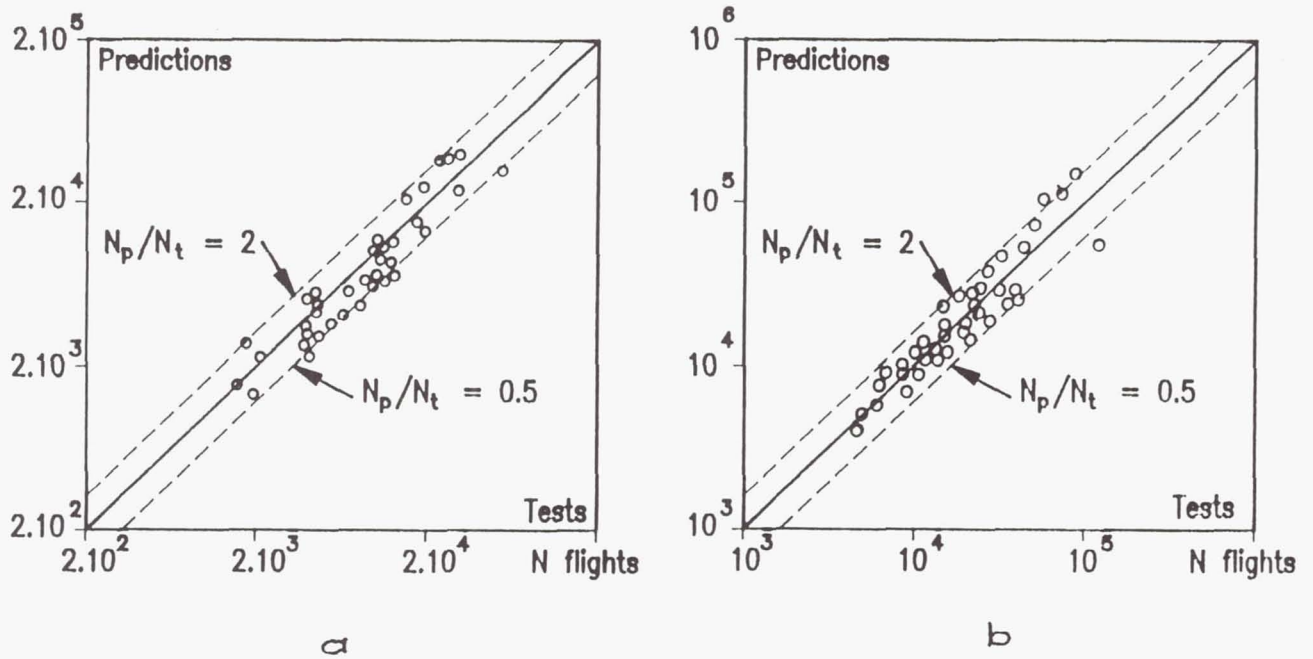


Fig 14 - Crack growth predictions vs tests for :  
a) - FALSTAFF spectrum  
b) - F27 spectrum

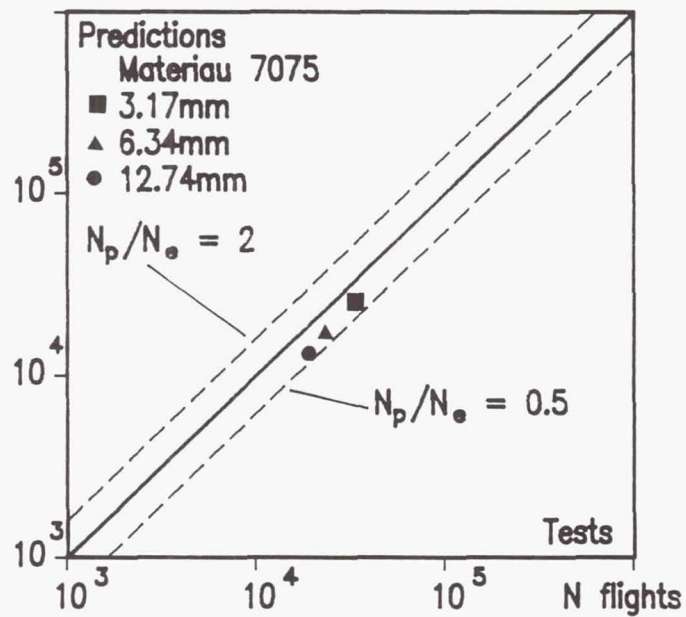


Fig 15 - Crack growth predictions vs tests for wing-upper-surface type of spectrum

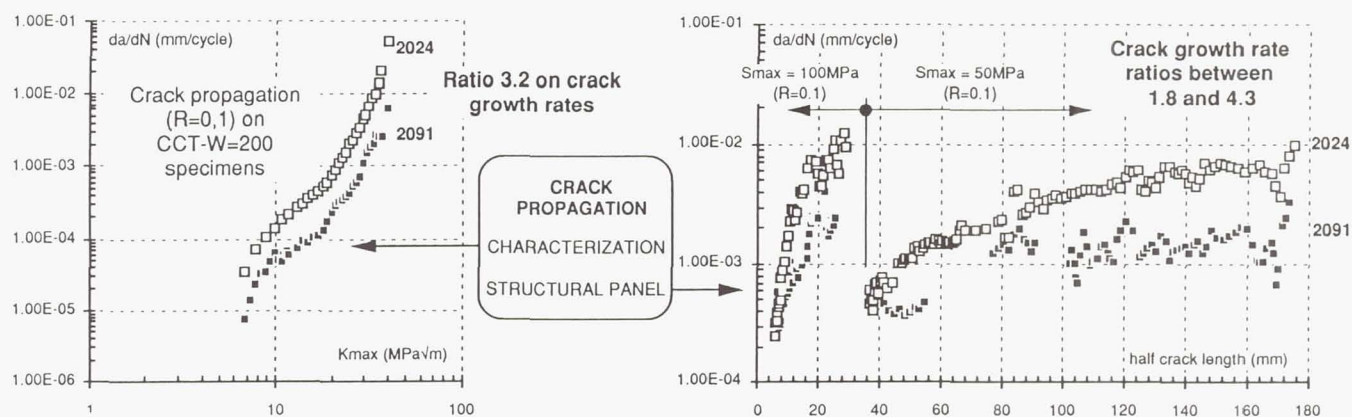


Fig 16 - Al-Li exploratory development, damage tolerance tests on typical fuselage segments : Crack growth aspects

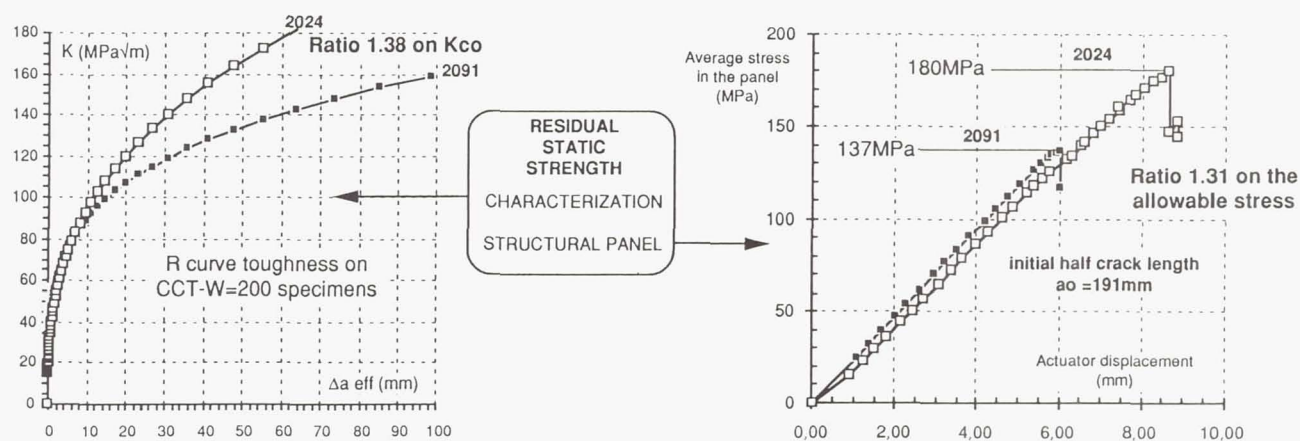


Fig 17 - Al-Li exploratory development, damage tolerance tests on typical fuselage segments : Residual strength aspects

Uniform compression loading in the sample



Local buckling of delaminated plies on the non-impacted side of the sample for a critical compression load



Reloading (compression) of delaminated plies located above the blister



Failure of the sample

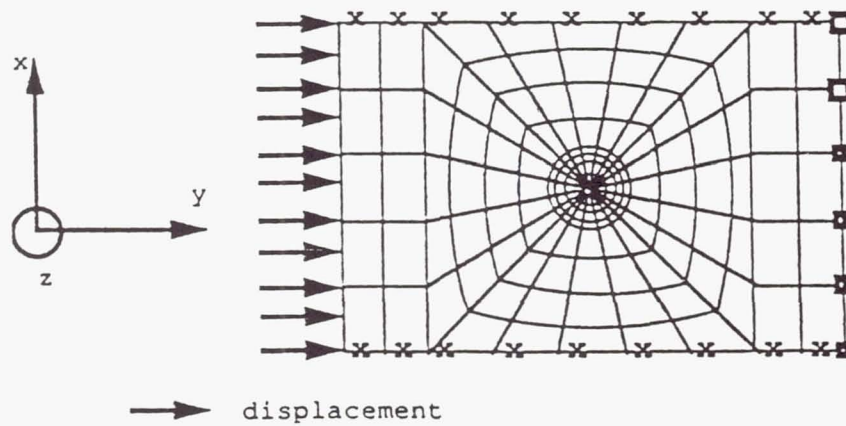


Fig 18 - Ruin mechanism and F.E mesh of an impacted composite specimen



# **A 310** **ENSEMBLE Tr15**

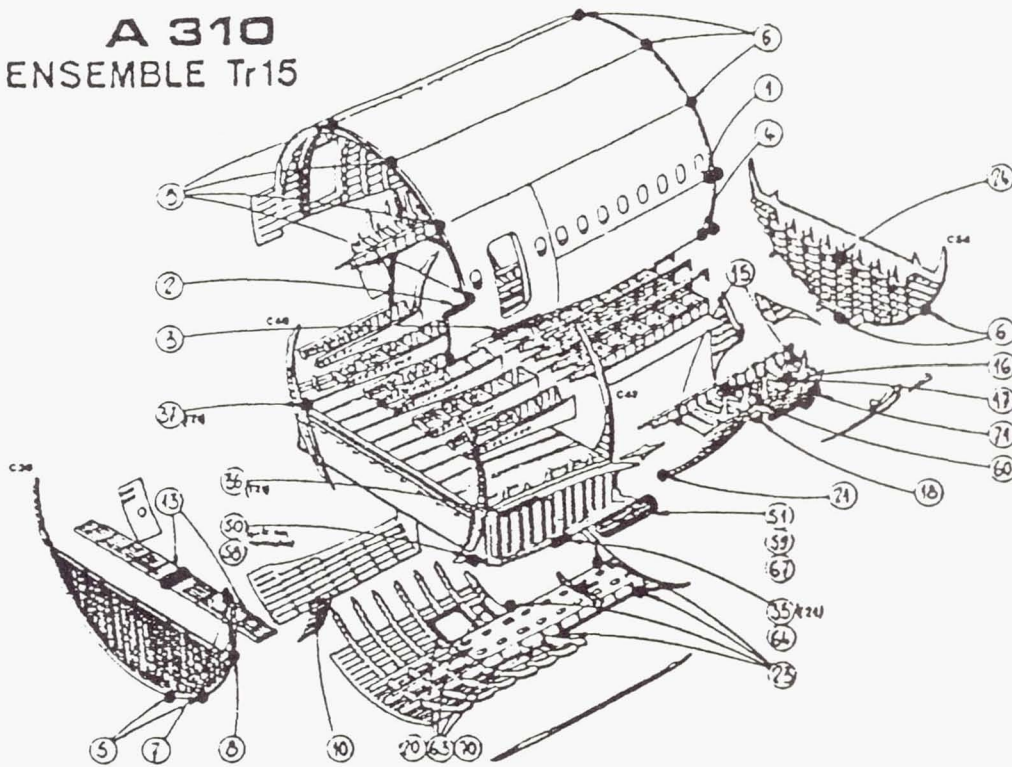


Fig 19 - Tear down - A310 center fuselage  
A310 ENSEMBLE TRONÇON 21

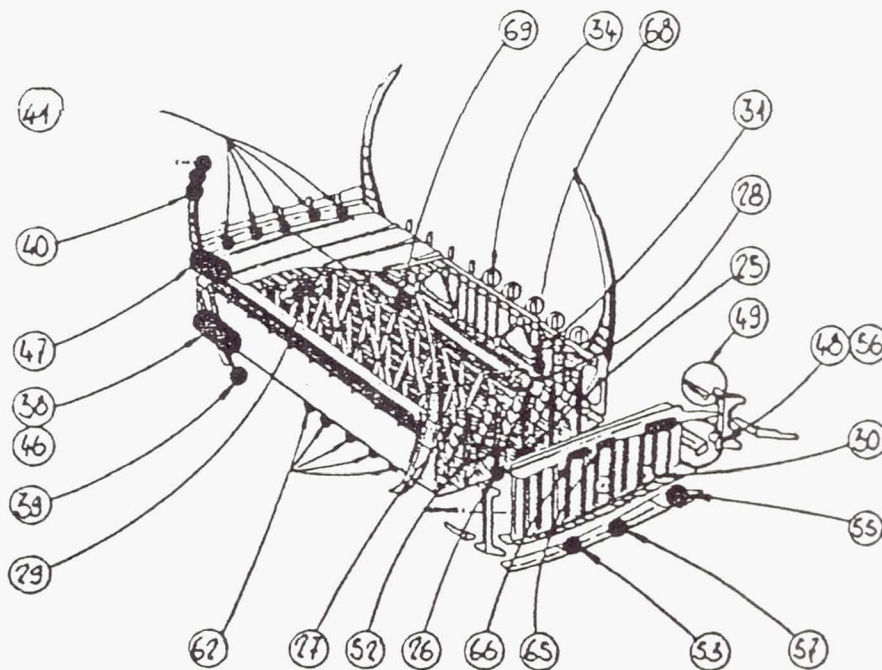


Fig 20 - Tear down - A310 center wing box

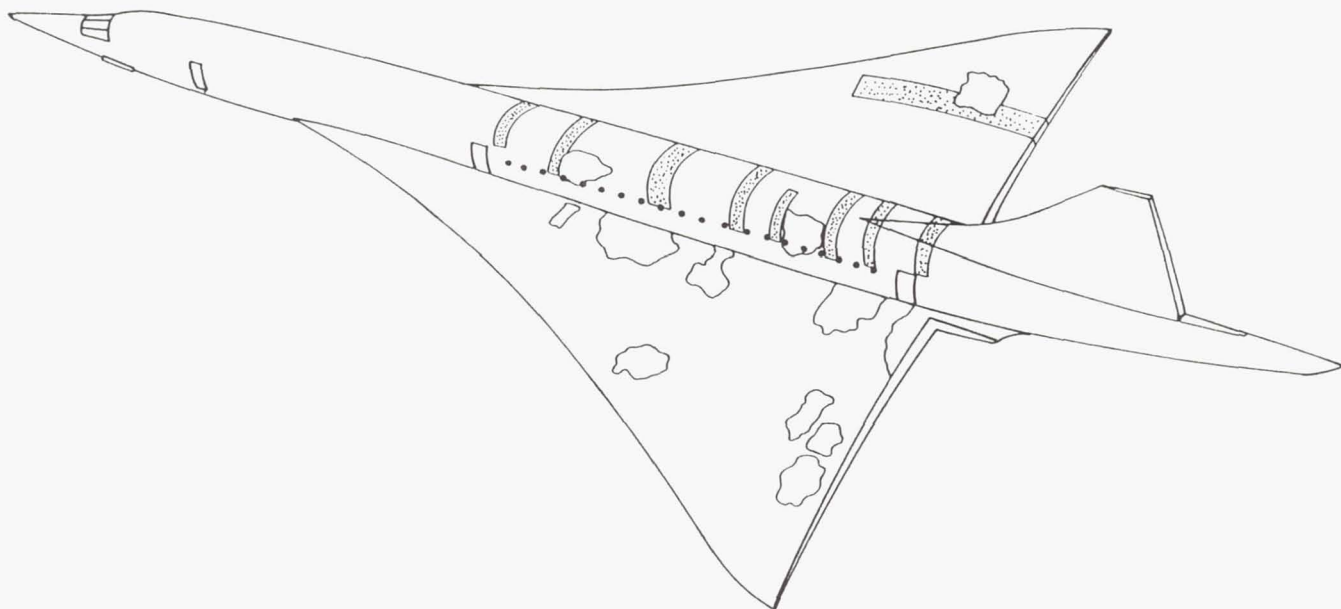


Fig 21 - Tear down - Concorde (locations of investigated areas).

TRANSPORT CANADA  
AGING AIRCRAFT ACTIVITIES

S.R. Didrikson  
Chief, Continuing Airworthiness  
Department of Transport, Canada

SUMMARY

A description is provided of recent initiatives undertaken in Canada to address problems of aging of passenger airplanes. In addition to participation in and support of US aging aircraft programs, independent activities have been undertaken in such areas as regulatory control of nondestructive testing, aging fleet evaluations and measures to address the airworthiness of aging Canadian-manufactured airplanes.

INTRODUCTION

In recent years the Airworthiness Branch of the Transport Canada Aviation (TCA) Administration has engaged in a program of activities to improve the airworthiness of older aircraft. This program has included activities specific to Canadian conditions as well as activities in support of or in parallel with the aging aircraft programs originating in the United States. TCA aging aircraft activities have been concentrated in the following areas:

- Measures to improve regulatory control of aircraft nondestructive testing (NDT);
- Aging fleet evaluation projects;
- Measures to address the airworthiness of aging Canadian-manufactured passenger aircraft, in accordance with the recommendations of the International Conference on Aging Commuter Aircraft held in Kansas City, in April, 1989; and
- Other related activities.

NONDESTRUCTIVE TESTING

On May 11, 1987, a 43 year old Douglas DC-3 crashed in Northern Ontario as a result of a wing structural failure. The accident investigation subsequently found that a radiographic inspection of the failure area prior to the accident had failed to detect a fatigue crack.

Canada has a system of certification of NDT technicians to standards of the Canadian General Standards Board (CGSB). Nevertheless, in view of the accident findings, TCA conducted a review of civil aircraft NDT procedures and technician licensing. A report was issued in January, 1988, which recommended a number of measures to improve regulatory control. These recommendations have been adopted and are being implemented as follows:



## TCA Regulation of NDT

It is proposed that all NDT will be brought directly under the regulatory control of Transport Canada by amendments to TCA airworthiness regulations to include it as "specialized work". This will require the NDT to be performed within the Approved Maintenance Organization structure with personnel qualified to CGSB or military standards and subject to TCA regulatory control.

Amendments to Transport Canada regulations and associated advisory material have been drafted and consultation with the Canadian aviation community is in progress.

## Maintenance Personnel Training

Steps have been taken to improve the NDT knowledge of aviation maintenance personnel by upgrading the NDT content in the curricula of accredited community colleges involved in Aircraft Maintenance Engineer (AME) training. In addition, AME examinations have been reviewed and revised to include questions that deal specifically with NDT.

## Transport Canada NDT expertise

TCA NDT expertise has been improved by the staffing of specialists in NDT. Also, NDT training has been provided for airworthiness personnel.

## AGING FLEET EVALUATIONS

Two aging fleet evaluation projects have been carried out, a Supplemental Inspection Program implementation review and an Aging Aircraft Sampling Evaluation project for large passenger airplanes:

### Supplemental Inspection Program Implementation Review

Following the Aloha Accident, a review team was formed and tasked to visit every Canadian air carrier operating Supplemental Inspection Document (SID) affected aircraft to assess implementation of the SID requirements. The team completed their review of the 18 Canadian air carriers affected and issued the final report in December, 1988. They were able to confirm compliance with the SID requirements by all affected Canadian operators. They found some weaknesses in TCA surveillance of air carrier SID implementation, which have since been corrected. The review also served to confirm the need to implement the recommendations of the NDT study.

### Aging Aircraft Sampling Evaluation

TCA Airworthiness Inspection staff were invited by their counterparts in the US Federal Aviation Administration (FAA) to participate in some of the FAA Aging Fleet Program evaluation activities. The appreciation gained of the value of these activities influenced us to develop a parallel program in Canada, which was called

the Aging Aircraft Sampling Evaluation.

The survey of the condition of older passenger aircraft in service in Canada was carried out between August, 1989 and May, 1990, by visits to repair & overhaul facilities to observe high time aircraft undergoing heavy maintenance. The objective was to obtain first hand knowledge of the condition of a sample of older Canadian aircraft, and to assess the maintenance systems that support them, in order to determine any corrective action required.

The aircraft surveyed were a Boeing 727, B-737, McDonnell Douglas DC-9, DC-10, Convair 580 and Hawker Siddeley 748. Most of the aircraft were in relatively good condition considering their age. The exception was the B-727, which had recently been imported, and was extensively corroded.

The final report of the Aging Aircraft Sampling Evaluation team included 28 findings and recommendations, concerning such matters as corrosion control and prevention, repairs, nondestructive testing, and imported and leased aircraft. The recommendations of the report were assessed and adopted and are being implemented.

Presently we are seeking funding to carry out a similar evaluation of aging Canadian commuter aircraft.

#### CANADIAN AGING COMMUTER AIRCRAFT

The International Conference on Aging Commuter Aircraft held in Kansas City in April, 1989, included in its review of aging commuter airplanes the Boeing of Canada Ltd, de Havilland Division DHC-6 and DHC-7, which were manufactured in Canada. Following the conference a summary report (ref. 1) was issued, which included the following recommendations:

- Existing airworthiness directives (ADs) on all airplanes used in regional air carrier service should be reviewed to determine if repetitive inspections need to be replaced by terminating actions;
- Existing service bulletins (SBs) should be reviewed to identify issues specifically relating to aging airplanes, with the intent of possible mandatory compliance;
- Corrosion manuals and related documents should be prepared or updated; and
- Supplemental Inspection Documents should be developed, issued and mandated by ADs.

Following release of the conference report, Boeing of Canada Ltd, de Havilland Division (DHC) were requested to organize airworthiness directive and service bulletin reviews. This was done, with the participation of operators, the FAA and Transport Canada.

The DHC-6 and DHC-7 AD/SB reviews were held in April and June, 1990, respectively. For the DHC-6, ten ADs were selected for terminating action and four SBs were selected for AD action. For the DHC-7, three ADs were recommended for terminating action and seven SBs deemed to warrant mandating by ADs. TCA have issued airworthiness directives to fully implement the recommendations of the

reviews.

DHC were also requested to develop corrosion manuals for the DHC-6 and DHC-7. Work is progressing satisfactorily and the manuals are expected to be completed early in 1992. Transport Canada expects to issue ADs to mandate the corrosion programs.

DHC have also been requested to develop SIDs for the DHC-6 and DHC-7. They have proposed instead that a retirement life should be fixed for the DHC-6. The engineering substantiation to support the life proposed is presently under discussion between DHC and TCA. DHC have also indicated their intention to develop a SID for the DHC-7.

#### OTHER RELATED ACTIVITIES

Transport Canada airworthiness engineers are participating in the working groups of the US Airworthiness Assurance Task Force. The airworthiness directives resulting from Task Force activities are being implemented in Canada.

At the request of Transport Canada, Bombardier Incorporated, Canadair Aerospace Group have developed a SID for the CL-44, a four-engine turbo-prop airplane manufactured in Canada. An AD has been issued to mandate the SID.

An amendment to the TCA regulations and an accompanying advisory document have been issued to give TCA the authority to require manufacturers to produce SIDs. This is equally applicable to imported and Canadian-manufactured airplanes.

#### CONCLUSIONS

The commercial operation of aircraft whose age exceeds their original design objectives is likely to persist indefinitely.

Much excellent work is being done by air carriers, manufacturers and airworthiness authorities worldwide to improve the safety of operations of aging airplanes.

Transport Canada support and cooperate with these initiatives. In addition, TCA are taking action:

- to ensure that Canadian regulations and maintenance systems address older aircraft (with particular emphasis on NDT);
- to assess the condition of older passenger aircraft operating in Canada; and
- to work with Canadian manufacturers to facilitate the support of aging Canadian-manufactured aircraft.



#### REFERENCES

1. Summary Report, Regional Airline Association, General Aviation Manufacturers Association International Conference on Aging Commuter Aircraft, April 25-27, 1989, Kansas City, Missouri, pages 48-50.

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# **Aging Commuter Aeroplanes - Fatigue Evaluation and Control Methods**

**A.J. Emmerson, Civil Aviation Authority of Australia**

## **Introduction**

The risks associated with old aeroplanes arise in the main from the difficulty of obtaining support from the manufacturer and other skilled people, and from a general loss of reliability caused by wear and other deterioration.

The loss of reliability is caused by two broad classes of problem. There are those problems which are self evident, and hazardous rather than catastrophic. These are the problem areas where characteristically there have been multiple overhauls, repairs, and replacements, and where aging really means the results of repair ineffectiveness that accumulates.

The other class of problem is the insidious and potentially catastrophic class. It includes the progressive deterioration of items which are not maintained, and often cannot be maintained because the deterioration cannot be seen. It includes loss of physical properties in adhesives and other organic compounds, corrosion, and the response to repeated loads. This paper deals with a currently unnecessarily troublesome aspect of that response.

The present working definition of "commuter aeroplanes" encompasses a wide range of aging types, from the Fairchild F27, through deHavilland Canada Twin Otters to the smaller twin engined Piper and Cessna models. It includes such types as the Lockheed 10, the Beech 18, and the Grumman Mallard. Although we must remain concerned about those types which have been certificated under a design standard or operational rule which embodies the elementary fail-safe concept and which have not been subjected to a subsequent structural audit, this paper is directed, in the first instance, towards types for which fatigue and damage tolerance evaluation was not required as a condition of certification. It will be seen, however, that the principles are robust enough for more general application.

## **An Australian Historical Perspective**

In Australia, the first recognised fatigue accident to what would now be called a commuter aeroplane happened on 31 January 1945. A Stinson Model A, the forerunner of the current range of twin engined Piper commuters, crashed enroute from Melbourne to Mildura. All ten people on board were killed.

The ensuing public inquiry recommended to the Australian Parliament that aircraft maintenance engineers be given special instructions covering the inspection of vital parts of



aeroplanes, that measures for non-destructive inspection should be instituted, and that the operational lives of aeroplanes should be limited.

Those recommendations were adopted, although the methods used were of course quite crude by today's standards. A method of evaluating the deterioration due to cyclic loads when airborne had only just become available (1-3). The lack of sophistication in fatigue evaluation was not evident at the time. But we were soon to learn. On 15 October 1951, deHavilland Dove VH-AQO crashed killing the seven people on board when the wing suffered a fatigue failure enroute from Perth to Kalgoorlie. The Dove on which the investigators travelled to the accident site was also found to have a large fatigue crack in the wing. VH-AQO and her four sister ships were the first Doves in airline service anywhere in the world. VH-AQO was four years old at the date of the accident. She had accumulated 9000 flying hours. Had the safe life been calculated by present methods, the wing of VH-AQO would have been retired after 2500 flying hours.

In the USA, the first explicit requirement for fatigue evaluation for general aviation aeroplanes was introduced with CAR 3 in 1957. At first only pressure cabins were covered, but in September 1969 Amendment 7 to FAR 23 required the evaluation of wing structure. FAR 23-7 required the wing to be evaluated on either a safe-life or a fail-safe basis. The fail-safe residual strength capability was 75% of limit load, to which a "dynamic factor" of 1.15 was to be applied.

In making this amendment, in August 1969, the FAA said:

"Service experience and discussions with industry of designs which have sustained fatigue failures indicate that present design practices do not adequately account for fatigue. Corrosion, prior abuse, and special purpose operations may contribute to fatigue failure, but the primary reason for such failures is lack of strength. Since fatigue failures are time dependent, a higher failure rate among older airplanes is to be expected; however, fatigue problems have arisen in airplanes certificated in recent years. Neither fatigue substantiation nor better maintenance programs will eliminate all cracks. The purpose of the proposed rule is to prevent catastrophic failures. Furthermore [FAR 23] airplanes are not designed on a redundant-structure, fail-safe basis for which maintenance alone would be sufficient. Both fatigue substantiation and a good maintenance program are needed (4)."

We believe the FAA was referring to a dozen or so types which had been designed to gain performance advantages from modern materials and design techniques, but which were certificated to the standard preceding FAR 23.

A major problem from Australia's viewpoint was that the new FAR 23 only applied to applications for certification of types completely new to the USA. Under the "grandfather clause" improved models of existing aircraft can be approved on the basis of the original Type Certificate. (To illustrate, all Mooney M20 series aeroplanes are covered by the same FAA Aircraft Specification 2A3, from the wooden winged M20 of 1955 to the 173 knot 2900 lb M20 K of 1982.) Thus there was a potential for a large range of problematical US aircraft to be imported into Australia for which no fatigue substantiation had been provided to the FAA.

Accordingly, in 1970, Australian airworthiness engineers visited the USA to discuss in detail with the FAA, and the manufacturers, an Australian proposal to require fatigue evaluation of all aircraft types entering Australia for the first time. The proposal was endorsed by the FAA. The USA manufacturers cooperated freely with evaluations of both new types and of older types already in Australian service.

The Australian airworthiness authority then imposed retirement lives on several of the types at risk. Those lives were based on:

- stress analysis and measurement
- knowledge of the fatigue resistance of similar classes of structure
- measurement of the severity of Australian service
- fatigue test data and lifing recommendations from the manufacturers
- failures in service
- Miner's linear cumulative damage rule (3)
- specified risk of failure of principal elements

The general evaluation methods used in Australia were very similar to those adopted by the FAA and the general aviation manufacturers in the USA as a standard approach in 1973. They have been published in FAA report AFS 120-73-2 Fatigue Evaluation of Wing and Associated Structure on Small Airplanes.

The Australian authority carried out a programme of measuring aircraft normal accelerations resulting from gusts and manoeuvres. Some 75 aeroplanes were fitted with recording accelerometers and a total of some 150,000 hours flying was recorded and analysed. The results are available from the author.

Control of the continuing airworthiness of these aircraft in the face of fatigue rested on five points:

- Thorough fatigue evaluation of the principal bending elements at the time the type is first certificated ( if the prerequisites for safety by inspection are not met, a retirement life must be set for the spars, unless the estimated safe life is greater than 40,000 hours)
- Mandatory defect reporting, close monitoring of overseas experience, and effective investigation of failures
- Spars may be changed, but primary structure which has not been evaluated may remain in service for only twice the life of the spars
- To extend the life of the aeroplane beyond two spar lifetimes there must be a structural audit and a review of the initial evaluation with a view to demonstrating damage tolerance of the principal structural elements up to some new extended retirement life



- Continuous amicable contact and free exchange of information between the operator, the manufacturer, and the authority is essential.

Retirement lives for aeroplane types vary between models and operational roles, and according to the confidence the airworthiness authority has in the evaluation. Table 1 is an indicative list of the lives now applied to some relevant types in Australia.

The anticipated outgrowth of the pre FAR 23 types did in fact occur. The general aviation twins now in commuter, air ambulance, and charter use are derivatives of earlier models and are frequently operated in a manner quite unforeseen by the manufacturer or the original certificating authority. Each year's model change of a few pounds weight, a few knots in speed and perhaps an extra two seats meant that the original basic Type Certificate could be extended. Ten or twenty years of this treatment produced aircraft having ideal performance for commuter passenger carriage with utilisation up to 2000 hours per year.

The Swearingen SA 26AT Merlin II is a good example of such a growth aeroplane - although it happens to have its own Type Certificate dating from May 1956. The Merlin II was manufactured by using wings and other components from used Beech 50 Twin Bonanza aircraft (certificated in 1949), and fitting a new pressure cabin and turbine engines. The gross weight rose from 5500lb to 10000lb, the engine power from 240HP to 972HP and the maximum cruise speed from 157 knots to 208 knots. Apart from some pressure cabin work, no fatigue evaluation was required or supplied for United States certification.

In Australia a life of 7500 hours was applied to the wing of the Merlin II. Similar action was taken with the Queen Air and King Air which were themselves growth versions of the Beech 50 using the same wing spar. A wing failure of a Beech 90 King Air in Canada and numerous spar fatigue cracks discovered in other Queen Air and King Air aeroplanes lend weight to the validity of the Australian action which had been vehemently opposed at the time. Beech aircraft are not alone of course in experiencing wing cracking. AeroCommander and Cessna and Partenavia twins, for example, have also experienced major cracks in main spars caused by fatigue. A fairly complete summary of accidents caused by fatigue in primary structure is in (5).

In an attempt to introduce some retroactivity into the requirements, Amendment 18 was added to FAR 135 in July 1970. FAR 135.169 carries a fatigue evaluation requirement for aircraft operating commercially after 1972 and having ten or more passenger seats. The FAA introduced Special Federal Aviation Regulation No. 41 in October 1979 to permit certain FAR 23 aircraft with more than 19 passenger seats to operate at higher than normal weights. A condition of the weight increase was that additional fatigue substantiation was to be provided, covering the vertical fin, horizontal stabilizer, and attaching structure. When SFAR 41 expired in 1989, FAR 23 was amended to include a commuter class with the same fatigue evaluation requirement. Australian and American certification requirements are now essentially identical for this class of aeroplane.

The current situation is that some 75 or so aircraft types have retirement lives placed on their structure in Australia. Of those, the retirement life is endorsed by the country of origin's airworthiness authority in all but eleven types. Those eleven types include eight pre 1969 twin engined types. They are no longer in production, they are getting old, they have a high and severe usage in Australia, and most individual aeroplanes are approaching their retirement life. They are nevertheless aeroplanes of some economic importance.



We have about 1250 light twins in passenger carrying operations in Australia. The Cessna 400 and the Piper PA 31 typify the types used in charter and commuter operations. There are some 200 Cessna 400 aeroplanes ( and 160 Cessna 300 which share the same wing design ) plus 210 Piper PA 31 in Australia. Most of them are in commercial service.

The service history of commuter aeroplanes presents a problem for regulatory authorities. There has been a significant number of fatal accidents caused by fatigue failure in the primary structure of that sort of aeroplane - aeroplanes of the same general class of design and construction, in the secondary airline operational and maintenance environment. On the other hand, there seems to have been a large number of aeroplanes of the class flying over the years that ought to have given confidence in their durability. Table 2 illustrates. But, to be truthful, we do not know how many of these aircraft are carrying cracks that have reduced their strength to less than the requirements of the design standard.

It is possible to evaluate this fleet experience. However the evaluation cannot be based on anecdotes. Hard evidence is required of aircraft that are structurally identical, and that have been flown in similar ways, and that have been free of defects.

Three final observations should be made about this Australian retrospective. Firstly, retirements due to the aging process have not just been confined to metallic aircraft structures. In the late 1950s several "commuter" types with glued wooden spars were retired from service because of progressive deterioration. They included for example the Avro Anson and some Percival types. The lesson is there for fibre composites.

Secondly, if in a fleet the cost is reckoned as the cost of replacing spars, and the benefit is reckoned as the saving of ten lives in the first accident, the benefits of Australia's policy have clearly outweighed the costs. Difficulty arises of course when the obvious cost accrues to the operator and the invisible benefit accrues to the public.

Thirdly, the application of retirement lives has been unduly prominent because of the lack of damage tolerance in many types, and of course as a result of the commercial implications of retirement. Since Australia embarked formally on the control of fatigue in civilian aircraft in 1945, the preferred method of control has been to maintain safety by inspection.

A review of the literature will show Australian contributions among the earliest papers on safety by inspection - for example Shaw's paper of 1954 (6). More recently, the CAA has encouraged and assisted local organisations to undertake damage tolerance evaluation of commuter types, and to design and fit modifications that will convey some measure of damage tolerance so as to permit safety by inspection. The Nord 262, in its Frakes 298 form, the Cessna 400 and the Piper PA 31 are the more significant types tackled successfully.

It is interesting, if not surprising, in this work how often the obvious is overlooked. Despite the commercial penalty of poor damage tolerance in some commuter types, no one in the Australian industry has yet proposed that on the expiration of safe life, 7075-T6 spar caps should be replaced with 2024-T3 caps of slightly greater thickness.

## Evaluation and Control Methods

### The Safe Life Concept

Before we can accept that an aircraft's structure can be kept safe by inspection we must know where to look, we must be able to look there, and there must be time to look. We need to determine how to look and when to look. The implications of accessibility and slow crack growth, the ability to tolerate damage, are clear. If these criteria cannot be satisfied, there is no airworthiness control alternative but to plan to retire the aircraft from service at some safe life.

The history of fatigue evaluation has in that way caused "safe-life" to be regarded as synonymous with "retirement life" and with retirement as a control technique; the adjective "safe-life" is also commonly used to describe a class of structures that should be controlled by retirement. None of those associations are necessary, and in hindsight we have been a little careless with our terminology.

To start with, a safe life only becomes a retirement life if it is promulgated as such by the airworthiness authority. But more generally, we should recognise that *a safe life is simply the life at which the risk of failure is predicted*, in the absence of information to the contrary, *to become unacceptable*. It is the time by which we had better be doing something.

Depending upon our control strategy, the safe life may be either the latest time by which we should have retired the aircraft, or the time by which we should have begun inspecting the structure for cracks if we are going to maintain safety by inspection.

In either case *the safe life will be the time at which the chance that the structure is unable to carry ultimate design load has become unacceptable*.

### Methods of Estimation

Consider for a moment the way in which this loss of strength occurs. There can be said to be three phases in the generation of cracks due to repeated loads. The first is one in which so far unobservable changes are taking place on a sub-crystal-lattice scale due to an unknown process about which there have been some metallurgical postulates.

The second phase is that transition phase which starts when the micro structure changes caused by repeated loads are above lattice scale but are too small to be detected or measured by conventional means, and for which the applicability of the concepts of fracture mechanics has not been demonstrated and is theoretically dubious because the material is not a continuum on this scale.

The third phase is the growth of identifiable cracks - a phase in which our rules of stress intensity and crack growth rates have been shown to apply.



It is not at all clear at what point strength first falls below virgin strength but it is conventional to associate this with the limit of resolution of nondestructive inspection - at the end of the transition phase.

Phase one changes and phase two changes are localised and accelerated by the presence of cracklike defects left during manufacture, and, equally, by the presence of micro structure irregularities of a metallurgical scale. The description of the fabricated material for analytical purposes must include an accounting for the distribution of flaws of these various sizes. Historically this has more often been implicit than explicit.

With sufficient accuracy for the purpose, in the class of structures we are considering, the conclusion of the transition phase and the point at which the strength falls below ultimate load capability may be regarded as coincident. Few of these structures could withstand ultimate load in the presence of the stress concentrating effect of a crack. The end of the transition phase is thus reasonably regarded as coincident with the safe life.

There are in use two methods of estimating this life - the linear cumulative damage hypothesis, and the growth of initial flaws hypothesis. It may not be too adventurous to proffer the opinion that this dichotomy is impeding the development of control methods for aging commuter aircraft.

It has been the author's observation over the past fifteen or so years that the proponents of these two methods have become implacably resistant to one another's views. This seems to be because:

- The need to have resistance to damage or flaws cannot be denied and thus there must be some study of the propagation of less than obvious flaws, and such flaws could be present just after manufacture.
- The methods have in the past been associated with allegedly undesirable or inappropriate practices; and,
- The essential similarity of the two methods has not been widely recognised.

It would be remarkable if that last proposition were to go unchallenged. But consider the matter more broadly.

Through nothing more than the sequence of development in the history of fatigue evaluation, a quite unnecessary connection has been made between the initially unreliable linear cumulative damage hypothesis and the sometimes wasteful imposition of retirement lives. The two are not congruent. We can have one without the other.

Similarly a connection has been made between the less explored propagation of initial flaws hypothesis and the safety by inspection process. The two have crack growth calculations in common but they are not congruent and not necessarily conjugate concepts.

In linear cumulative damage analysis the following steps are used:



- Begin with data describing the endurance under repeated loads of structure of the general type for the member being evaluated - usually in the form of an SN curve established by pooling full scale test data.
- Scale or calibrate that curve so that when using Miner's cumulative damage rule the curve predicts the results achieved by a fatigue test of the actual component.
- Apply the anticipated load spectrum, as stress range pairs, to the calibrated curve to estimate mean life in service.
- Using standard statistical techniques, correct this estimate for a want of confidence in the mean life; and, using Impellezziri's proposition that the variance between nominally identical structures is a characteristic of the class of fabrication (7), calculate the safe life.

This technique, although derived from Miner's 1945 hypothesis, has benefited from sophistication added over some thirty years - principally by the members of the International Committee on Aeronautical Fatigue. It would be better known as the Miner/ICAF method.

The propagation of initial flaws approach uses the following steps:

- Begin with data describing, for structure of the general type of the member being evaluated, the range of initial flaw sizes and their frequency of occurrence. Choose an initial flaw size, the frequency of occurrence of which corresponds to the acceptable risk of failure.
- Develop endurance data in the form of crack growth curves for particular locations from standard curve shapes calibrated to replicate the results of crack growth measurement in tests of the member being evaluated. Correct that data for want of confidence in the test result representing the mean.
- Apply the anticipated load spectrum, as stress range pairs, to the calibrated curve to estimate safe life in service.
- Repeat this process for other crack locations in the same member.

Although this method arose out of the F111 recovery program, the crack-like initial flaws postulate was made by Griffith many years earlier. As the crack growth equations stem from the initial work of Paris in 1964 (8), the method might be called the Griffith/Paris method.

Several points should be made about these techniques. The original Miner method gained a reputation for unreliability. This was very largely a result of the pitfalls that were at the time undiscovered. The principle lessons over the years have been the extent of the innate variance between nominally identical structures; and the need for SN data that is thoroughly representative of the structure being evaluated and its propensity to develop cracks. The SN data must account for poor design geometry, built-in stresses, fretting, and the stresses due to incidental loads.

Some pitfalls have also been discovered in the Griffith/Paris method. It is fair to argue that as yet the technique is not mature, and that there has not been sufficient application to confirm any suspected unreliability. In commuter aircraft, we have seen very few failures from

crack-like manufacturing faults of discernible size, other than bad welds and hydrogen embrittlement.

SN data drawn from full scale tests most often covers a fair range of failure modes in the one curve. Crack growth curves are particular to a location and this adds to the computational work.

The essential elements of the two methods are the same. Externalise the variability between nominally identical items. Calibrate an empirical deterioration curve representative of the class of structure against a full scale test. Then by piecewise integration calculate the number of flights to reduce strength to a specified value.

If we were to search for a difference in the physics implied by the two methods, an important theoretical difference would be that the endurance curves are of a different nature. In one the estimated microstructure response to load cycles is independent of what has gone before. In the other the response is dependent on what has gone before. The question that might be asked then is what are the physics during the safe life of commuter structures. The best answer is that while crack sizes remain small, and providing the curves are properly calibrated, it doesn't matter.

If there is a significant difference in the reliability of the methods, it will arise from the extent to which the final endurance data was derived from experiments in which the distribution of initial flaws, micro or macro, or propensity to crack, was the same as in the structure being investigated.

If the author were asked to express a preference for evaluating small commuter aeroplanes, it would be to use Miner/ICAF to safe life and fracture mechanics beyond that - simply on the grounds of computational simplicity and the availability of data. There can be no objection to a Griffith/Paris approach well based on measured data.

Endurance data should be calibrated against the fleet as a flying fatigue test. The desk top computer has made this perfectly practicable - along with full allowance for variability in the combined fatigue and crack growth processes. It is probably now too late to worry about the probability of early failures from the largest manufacturing flaw which might have escaped detection.

To end on a philosophical note,

"It seems that no other scientific area has been studied so intensely and yet resulted in so little correlation between quantitative empirical data and theoretical computations based upon some hypothetical physical mechanism. The underlying reason for this lack of consistent correlation is the complexity of the basic mechanism of fatigue. The fact that the fatigue of metals takes place on a submicroscopic scale requires highly sophisticated testing techniques in order to observe the physics of the phenomenon [and in particular] what is essential to the mechanism and what is incidental (7) ."

Engineers ought not to rely on entirely self referential doctrine - doctrine which has its fixed points of reference inside itself. Such doctrine has an instability analogous to a statically indeterminate mechanism. Our theory must be pinned at least at one end on irrefutable external



evidence. The position of the other end can then be located by inference. The physics of the response to repeated loads, preceding visible crack growth, have not been directly observed. The Miner/ICAF theory and the Griffith/Paris theory are in the first instance self referential theories.

The aging aircraft issue is topical but it is not a new issue. Aging aircraft have periodically caused concern throughout the seventy year history of Australian commercial aviation. While this may have been especially so in times of economic difficulty and undercapitalised airlines, Australia's good years produced utilisation rates that frequently made Australian aircraft the leaders of the world's fleet. The particular problem of structural aging or fatigue is one in which Australia has perforce had a special interest over the years. It is a particular area where regulatory interest by Government is necessary to achieve a proper balance between the pressures of commerce, and public safety in the long run.



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Table 1 Retirement Lives of Some Components of  
Representative Light Twins

Aircraft		Component	Retirement Life (hours)
Cessna	401/402	Wing main spar lower cap	8200
	402C		7700
	421C (early)		6400
	421C (late)		17300
Piper	PA 31	Wing main spar lower cap	11000
	PA 31-350		13000
	PA 42-1000	fuselage empennage	13650 cycles 37500
Beech	65 B80	wing main spar lower cap	12000

Table 2 Production History of Representative Light Twins

	Model	First Flown	Number Built
Cessna	411	1962	301
	401	1966	404
	402	1966	1540
	404	1975	378
	414	1968	1067
	421	1965	1909
	425	1978	232
	441	1975	360
Piper	PA 31	1964	800
	PA 31-350	1978	1850

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The base for structural integrity and high reliability is set already at the design stage by sound engineering design concepts based upon requirements of integrity, simplicity and maintainability. Important for high reliability is internal load distribution, fatigue resistant materials, high quality joint technologies and verified good corrosion protection systems. Very important and basic to all design is knowledge about conditions under which the aircraft are going to be used - load levels and sequences, environmental conditions etc.

It is thus, for the developer, important to have high quality engineering design and sizing methods. Important in such a concept are methods to obtain internal loads distribution based upon the knowledge of global load levels and sequences, methods for sizing to static, fatigue and damage tolerance criteria. Access to design allowables for the different materials under actual environmental conditions is also important.

Included in this concept are also the verification procedures where the design is verified by testing at different levels of complexity from simple coupon tests to fatigue and damage tolerance testing of a complete airframe. Verification of loads on the aircraft is made by flight testing and feedback from the usage of the aircraft. Internal loads are verified also by testing.

Quality assurance in production by qualification of processes and equipment and control of processes and details is important to secure the final product quality. Quality control in production by non-destructive testing and inspection methods is important, especially when using new advanced technologies like composite designs, bonding technology etc.

To secure high quality throughout the service life of the aircraft it is necessary to have a well developed structural maintenance program. This shall include analysis of the structure and classify structural significant items and fatigue, environmental and accidental damage analysis. From this analysis a structural inspection programme is developed.

In order for the customer to be able to maintain structural integrity and reliability it is important to develop a structural repair manual including descriptive information and specific instructions, material identification, allowable damage criteria and repair design solutions together with procedures and processes. This manual should be continuously updated with new repair schemes. In addition the customer should be supported by a product support team at the developer for repair and special damages that are not covered by the repair manual.

Feedback from the customers on how the aircraft are used and feedback of damage information to the developer is of utmost importance for being able to maintain structural reliability and also to improve future product quality by influencing future design solutions by experiences from the field.

Last but not least it is important that the operator develop and maintain high quality and skilled personnel that operate and maintain the aircraft according to defined procedures.

## **2. DESIGN CONCEPT OF SAAB 340 AND SAAB 2000**

Aircraft structure should be designed for a low risk of fatigue crack initiation throughout the design life. However it can not be guaranteed that fatigue cracking will not occur in at least some aircraft of the fleet. Aircraft structure may also be cracked due to corrosion, incorrect maintenance or impact damages from accidents such as turbine disk disintegration or propeller break-up.

This means that besides an overall high fatigue quality and a good corrosion prevention system, the structure must be designed also with damage tolerance capabilities. Such a damage tolerant structure should maintain sufficient residual strength until any cracks or damages are detected during scheduled inspections.

### **2.1 The structural design concept**

Saab's on-going programmes dedicated to research and development of new structural concepts, together with experience gained on military and other civil projects, provide a firm base of advanced technology on which the Saab 340 and Saab 2000 structural design are built. The structural design is based on three main objectives:

- o Integrity
- o Simplicity
- o Maintainability

The Saab 340/2000 makes extensive use of the PABST bonding technology in the fuselage and the wings, and of advanced materials on control surfaces. Structural simplicity is greatly enhanced by the use of bonding in more than 40% of the structure.

#### **2.1.1 The fuselage**

The fuselage structure consists of three major assemblies:

- o The forward section (with cockpit)
- o The centre section (cabin)
- o The rear section including cargo compartment

The forward and rear skin are partly a conventional design which makes use of chemical milling techniques and partly bonded design.

For Saab 340 and Saab 2000 the centre cabin section is built up of top, bottom and two side panels. These single-piece panels consist of an outer skin to which the doublers and stringers are bonded. Additional for Saab 2000 is that the top and side panels are spliced fore and aft of the centre section due to its length. Plug type doors are standard throughout the pressurized cabin.

### **2.1.2 The wing**

The wing is an assembly of left- and right-hand wing panels spliced at the aircraft centre line. As this is the only splice for these major components, good weight efficiency and high durability are achieved. The wing box structure is of conventional design built on two spars, with the volume between the spars over a certain wingspan used for an integral fuel tank with access doors in the upper panel. The upper and lower wing panels are bonded assemblies. Trailing edge panels are made of composite materials.

### **2.1.3 The empennage and control surfaces**

Sandwich panels with metal honeycomb are utilized in the construction of the fin and horizontal stabilizer. Bonded assemblies of honeycomb are used in order to reduce the weight and the number of parts. Furthermore, these structural techniques provide a smooth surface finish and low drag for further gains in performance efficiency.

## **2.2 The use of structural bonding**

Saab Aircraft started to use adhesive bonding on the Saab 29 fighter aircraft "The Flying Barrel" in production in 1953 and has applied this technique successfully over time to an increased extent in primary structures of the aircraft Saab 32 "Lansen", Saab 35 "Draken", Saab 105, Saab 37 "Viggen" and Saab JAS 39 "Gripen".

Rationalization for metal to metal bonding are:

- o Weight optimal
- o Damage tolerant structure
- o Better aerodynamic performance due to smoother surface
- o Increased stiffness in thin gauges design
- o Reduced needs for using mechanical fastener holes which are potential candidates for both initiation of fatigue cracks and corrosion
- o Inherent corrosion protection in areas with doublers

Saab has developed the bonding technology to bond large complicated panels with skin, doublers, stringers, reinforcement frames and window frames in one or more bonding operations. Metal to metal bonding is sometimes used even for double curvature panels. Figure 1 shows a metal to metal bonded structure.

## **2.3 The corrosion protection system**

Corrosion protection of the aircraft is among the most comprehensive in industry. A fully automated facility ensures the anodization of the different parts before assembly. This is followed by applying a primer which ensures protection against the various forms of corrosion over a long period. The corrosion protection scheme is shown in figure 2.



### 2.3.1 Painted exterior surfaces

Our approach to make inspection easier of the exterior is to have a semi-strippable paint scheme. This means that we have chosen a system easy strippable down to a transparent organic film. The remaining coatings consist of phosphoric acid anodizing and a transparent bonding primer. There are a number of advantages with this scheme over a conventional one:

- o No grinding operation is required
- o A relatively mild non-phenolic stripper which does not degrade the adhesively bonded joints can be used.
- o There is no need for acid deoxidizing and swabanodizing of the stripped aircraft prior to repainting. Referred treatments may cause corrosion if residues are left in crevices. Good adhesion is secured by the bonding primer
- o The adhesion between the bonding primer and the phosphoric acid anodize is unsurpassed. It has been found that this pretreatment resists filiform corrosion better than any other treatments we have tested. Filiform corrosion occurs more readily on clad than on bare materials overcoated with polyurethane top coats. It has also been found that phosphoric acid anodize plus bonding primer gives better resistance against exfoliation corrosion around fastener heads than chromic acid anodizing does.

Hydraulic fluids are known to affect paint coatings. The hydraulic fluid based on phosphate ester called Skydrol is found to be the most effective paint stripper. A strippable paint scheme containing a wash primer does not resist Skydrol. The removal of paint schemes that resist Skydrol requires use of corrosive phenolic strippers, severe grinding operations and chemical brightening with an obvious risk for structural damage. Many airliners therefore reapply a strippable system on all exterior areas except for those known to be heavily contaminated by the hydraulic fluid. We have chosen to use a strippable paint system from the very beginning.

### 2.3.2 Protection of interior areas

There are a lot of crevices and fasteners in the interior area of an aircraft with a potential risk for corrosion. Our philosophy is to apply paint in the detail stage which also is desirable in securing good adhesion. During assembly, parts are fay and fillet sealed to standard praxis. After assembly of the whole aircraft but before installation of insulating blankets we apply a water displacing protective oil which effectively penetrates any crevice left. Several visits and interviews with airliners have indicated that such a protective oil is highly desirable during manufacturing of a new aircraft. It is now also required in IATA doc. 2637, issue 2.

### 2.3.3 Special design considerations

The design concepts are a determining factor for corrosion resistance. In order to get adequate protection, corners and edges on metals used in exterior locations or in areas exposed to a corrosive environment are broken prior to protection. This procedure is also applied to metals prior to shot peening. Items made of forging and plate product forms are protected as follows:

- o Cold working by shot peening or blasting
- o Polishing of critical radii
- o Chromic acid anodizing using a process with good reproducibility and painting.

Considerable condensation takes place especially inside the outer skin and in the ventilation system therefore provisions are made for adequately sized drainage paths. Contact between porous materials and metals is avoided. Thus, it is important that e.g. insulating blankets are not installed tight and wrapped around stringers or in contact with the skin.

All faying surfaces in exterior locations and corrosion prone areas are sealed unless not adhesively bonded. Special consideration is also taken for interfaces between dissimilar materials.

The use of magnesium alloys in aircraft design needs special care. We have some twenty-five parts of a cast alloy designated ZE 41A. These parts are protected by surface sealing with a stoved epoxy resin prior to painting and are not used in areas exposed for an aggressive environment. Whenever possible, this resin is applied also to tolerance surfaces in order to avoid direct metal-to-metal contact. The magnesium castings are also protected against galvanic corrosion by use of wet assembly practice, aluminium washers and sacrificial coatings on dissimilar metals.

Carbon fibre composite material also possesses potential risks for galvanic effects especially in fastener installations. This is restrained by use of fasteners of titanium, monel or corrosion resistant steels.

In areas where galvanic corrosion may occur attention should be paid to the fact that it is better to arrange for drainage from the less noble metal to the more noble one than reverse and that the anode area should be large relative to the cathode area.

In highly corrosive areas such as beneath the lavatory, plastic materials, corrosion resistant steel or titanium are used and the area is enclosed from other structures. Areas highly susceptible to corrosion are easily accessible to permit cleaning and inspection.

For metal sandwich structure it is important to restrict transportation of humidity into and within the panel. Square-edge designed panels with a non-perforated core type and with all edges and holes adequately sealed is our solution.

Additions to the fuel in order to suppress fungi growth in aircraft operating in tropic countries has been considered insufficient as a single means of avoiding corrosion. The integral fuel tanks are therefore protected by a fungi resistant coating per MIL-C-27725.

### **3. LOADS ANALYSIS**

#### **3.1 External loads analysis**

The external loads on the aircraft are analysed for all conditions specified in FAR/JAR regulations. In addition a number of states that occur during final assembly, jacking up during overhaul etc. are specified.

Determining the external loads requires analysis of e.g. structural dynamic response to specified extreme manouvres in flight, aerodynamic pressure distributions at different speeds, angles of attack etc. and gust loads, cabin pressurization, landing impact, taxiing, braking, turning on runway, etc.

Several different payload distributions are considered to find extreme loads for each state.

A large number of fail-safe conditions e.g. failure of certain structural components, failure of power assist devices, sudden engine stoppage, loss of propeller blade, etc. are also analysed for each state.

The load analysis makes use of techniques like the Finite Element (FE) method to predict dynamic transient response e.g. landing, Computational Fluid Dynamics (CFD) for prediction of aerodynamic pressure fields and 6 degree of freedom flight mechanics model with control system logics to predict in-flight manouvres. Numerical predictions are supported by wind-tunnel tests of models and finally verified during tests of complete aircraft.



### **3.2 Mission analysis**

The complete definition of external loads also includes the definition of the load sequence. The usage of the aircraft throughout its design life is defined in a hierarchical way.

Basic information used includes the definition of the expected flight plan which involves climb, cruise, descent, flight times, operational speeds and altitudes, and the approximate time to be spent in each of the operating regimes. Operations for crew training and other pertinent factors, such as the dynamic stress characteristics of the flexible structure excited by turbulence are also considered. For pressure cabins, the loading spectrum include the repeated application of the normal operating differential pressure, and the superimposed effects of flight loads and external aerodynamic pressures.

The definition and handling of this sequence of load cases are described in section 4.

### **3.3 Internal loads analysis**

The internal loads distribution in the structure is then obtained by solving the finite element model of the complete aircraft for all the defined unit load cases.

The present status of the basic loads analysis process is schematically shown in figure 3. Not shown in this scheme is the feedback from analysis to design and the verification process consisting of component tests, ground and flight tests.

## **4. FATIGUE AND DAMAGE TOLERANCE SIZING AND VERIFICATION**

### **4.1 The sizing and verification concept**

The design criteria for service life and damage tolerance requirements are met during the development phase through a fatigue and fracture mechanics concept based on what we refer to as the Global Spectrum Approach. This concept which is also a part of a global concept which involves applied regulations, feedback from structural and flight testing and follow-up of service experiences, etc. (figure 4). The method is computerized.

### **4.2 The computer aided system**

The computerized system is handled by any stress engineer involved in a project and is used for both sizing and test planning work. A pre-requisite for such fatigue and crack growth work is access to local load spectra or load sequences for any part of the structure. The information needed for this purpose are obtainable in each aircraft's two project central databases; these contain:

- o The design loading of the aircraft, defined as a sequence of instantaneous load cases defined for the complete aircraft.
- o Solved unit load cases for the finite element model of the complete airframe.

The system is in full production in both the military Saab JAS39 Gripen and the civil Saab 2000 aircraft programmes. Work is also going on to implement the whole system into a workstation based CAE system which involves graphical interfaces and transparent data communication links.

### **4.3 The loads spectrum handling procedure**

#### **4.3.1 Global sequence**

The design loading of the aircraft is defined as a sequence of instantaneous load cases defined for the complete aircraft and available in the global sequence.

The efficiency of the system to handle and process load cases is based on the hierarchic structure of information in the global sequence (figure 5). Three levels for load case definition and three levels for the composition of the load case sequence are available.

Balanced load cases are obtained from linear combinations of solved unit load cases for the finite element model of the complete airframe. The current global sequence for the Saab 2000 aircraft contains 7824 unique load cases which are composed from 224 solved unit cases. Every unique load case is also assigned a code number. These code numbers are used for identification and are particularly important for the preparation of the structural testing.

On the lowest level for sequence definition, parts of a mission are defined. Load case sequences for ground conditions consisting of turns, bumps, breaking, towing, etc., manoeuvre and gust conditions for different configurations of the aircraft and landing conditions are defined as subsequences. Having this bank of sub-sequences, a number of complete missions can be defined as sequences of these sub-sequences. At the top level, the mix of missions are defined. It is not necessary to define a unique sequence for the whole service life. A repetition sequence representing approximately 5 percent of the total service life is sufficient. For Saab 2000 a unique sequence of 3000 missions is defined. For the whole design service life of 75,000 flights the global sequence consists of more than 285 million states.

#### **4.3.2 Local spectrum**

Local sequences and spectra for any calculated quantity for any location in the finite element model can momentarily be created by the stress engineer through the use of the system. For complex situations the system has features to facilitate a number of operations on the finite element results. Loads, moments, stresses, displacements, etc. can be factorized and superimposed and principal stresses and effective stresses can be derived.

When testing is planned or when cycle-by-cycle crack growth prediction shall be made, it is sometimes advisable to omit certain small cycles which from a fatigue viewpoint have no practical meaning. Interactive facilities to remove those insignificant states are available. Rules and guidelines for choice of truncation levels are based on spectrum test results.

As an example, a finite element substructure representing a part of a wing attachment frame, skin and window for Saab 2000 is shown in figure 6. A local stress sequence for the marked portion of the inner flange of the frame is obtained. Figure 7 shows this local sequence after an omission operation. The omission technique adopted works also when more than one load sequence is considered through a lowest common denominator scheme.

For very long irregular sequences like this the rain-flow counting algorithm that in a relevant way combines states into cycles is available. The counting is performed on the subsequence level with intermediate states treated afterwards and has been proven to be very computer time efficient. The distribution of counted rain-flow ranges for the above described reduced sequence and the original sequence are shown in figure 8. The rain-flow matrix of associated peaks and troughs as well as the local stress sequence are linked to the fatigue and fracture mechanics processes of the system.



#### 4.4 The fatigue sizing procedure

The landing gear and their attachments are certified as safe-life structure and certification is based on fatigue testing. To minimize the risk of early failures in the tests and associated burden of modifications, thorough fatigue analyses are made.

Since all other structures are certified as damage tolerant, fatigue analysis is not a certification requirement. However, despite this fatigue, sizings are made for the following reasons:

- o To set maximum allowable stress levels, in order to reach a low risk for fatigue cracking throughout the design service life.
- o As one way to determine where, in a principal structural element, the assumed initial flaw for the damage tolerance evaluation shall be placed.
- o In order to avoid multiple site cracking within the service life. Areas particularly prone to this phenomena are e.g. long splices, lower wing panel at ribs and back-to-back fittings.

The system process used for this purpose is based on a cumulative fatigue calculation method of a "Relative Miner" type. This means that the allowable damage sums are based on spectrum test results and depend on the type of structural item and type of loading. Several testing based computer aided procedures for adjustments in order to account for various deviating conditions in applied cases, e.g. size effect, surface roughness, surface treatment etc., are available.

In the early phase of design, when the configuration is still open to changes, it is of great importance and generally advisable to study the effect on critical damage sum from a change in stress level and configuration. Several graphical aids for this purpose are available (figure 9).

#### 4.5 The damage tolerance sizing procedure

Damage tolerance means that the structure, at any time during the operational life shall maintain the required residual strength with an in-service detectable damage present, while subjected to the expected typical loading spectrum, until the damage is detected.

In the design and sizing of the structure it is therefore important not only to generate damage growth and residual strength data but also to consider the inspectability of the structure in-service.

To ensure residual strength in case of failure in an adhesively bonded joint some general and specific requirements exist. The bonded structure shall maintain the residual strength with any reasonably expected damage of a size that is likely to be detected by the in-service inspection program.

The principal structural elements and the damage critical areas where damages are simulated are identified accounting for the following factors:

- o Review of static analyses to locate areas of maximum stress and low margin of safety.
- o Select locations which by fatigue analysis are determined to be prone to fatigue.
- o Select locations where the crack path is short.
- o Select locations in an element where the stresses in the adjacent structure would be the maximum with the damage present.
- o Select locations where damage detection is difficult.
- o Select locations which, based on test or service experience on similar designs, are prone to fatigue damage.

For each critical damage location the most critical initial flaw is normally determined considering:

- o The initial flaw which will produce growth to a critical damage in the shortest time.

Considerations are also given to:

- o Initial flaws which will give the largest extent of damage before detection is possible.
- o Initial flaws which will lead to complete failure of one element in the shortest time.

Multiple or secondary initial flaws are assumed where the design is such that multiple site damages can be expected to occur due to fatigue.

The size and shape of the primary initial flaws are selected to cover manufacturing defects of what is generally known as a "rouge flawed aircraft". The secondary initial flaws are smaller since they are supposed to be caused by fatigue and subsequently in reality not existing at all on "day one" in the service life of the aircraft.

It should be pointed out that these initial flaws are not normally used to establish the inspection repeat intervals. The reason for introducing the primary flaw is to get a starting point for the crack growth prediction which accounts for manufacturing defects. The secondary flaw is used to determine the size of the damage for continuing crack growth.

The damage tolerance analyses are handled by fracture mechanics methods. The system process CAFE (Computer Aided Fracture Engineering) is doing this work and is linked to databases containing materials data and stress intensity factor solutions and has a broad range of methods and techniques to build and solve fracture mechanics models.

The principal result from the fracture mechanic analysis is a crack growth curve beginning from the assumed initial flaw size and ending at the critical crack size. This curve, figure 10, is used in the Maintenance Steering Group (MSG-3) procedure to establish at which time the normal in-service inspection programme for fatigue damages is initiated and the inspection repeat interval. The safety factors used to obtain the safe periods and intervals varies depending on the design.

#### 4.6 Structural test verification

The fatigue and damage tolerance sizing create the necessary conditions for high structural integrity. This integrity need also to be demonstrated in testing according to the sizing approach shown in figure 4. The tests are made in order to:

- o Verify freedom from premature fatigue cracks in primary load paths.
- o Identify any premature fatigue cracks in the structure.
- o Generate crack growth and residual strength information for artificial defects to correlate analytical predictions.
- o Verify calculated inspection periods for fail-safe structure in the fail-safe mode.
- o Determine inspection threshold and inspection repeat interval for naturally developed cracks.
- o Verify fatigue life for safe-life structure.
- o Verify freedom from multiple site damages where applicable.

The load spectrum handling process of the system takes here a principal part since it contains a large number of facilities which are necessary in the test preparation work task.

Besides the testing for obtaining data for predictions, e.g. fatigue curves for materials, lugs, joints, fastener systems and crack growth rates and fracture toughness etc, three main levels of testing are identified.

Detail testing is mainly performed early in the sizing work. It is used to verify detail design of vital structural members and to qualify the application of prediction methods to typical structural configurations.

Major component testing is done for early fatigue and damage tolerance verification. The key point in these tests is that a critical part is tested while properly installed in its nearest boundary structure. This type of testing will verify the structural integrity and the prediction of internal loads.

The final verification of the fatigue and damage tolerance performance is made with a complete airframe tested for several service lives. This test will spot fatigue critical areas and demonstrate the stable growth of those natural cracks that may initiate and of any artificially made cracks. The Saab 340 full-scale fatigue test has now exceeded 150,000 simulated flights. This test will continue to 180,000 flights before artificial cracks, of sizes which are in-service inspectable, are installed.

Examples of major fatigue and damage tolerance tests for Saab 2000 are given in figure 11.



## 5. QUALITY ASSURANCE IN PRODUCTION

The quality assurance of a product like a commercial aircraft is of utmost importance. The quality assurance task permeates all sub-moments in development and production. From a production point-of-view, the produced items must comply with the demands used in the design and sizing.

Flow diagrams for all work-shops are established and each sub-moment in this flow is supervised by instructions and quality control documents.

Some production processes can not always be verified by subsequent control and testing and therefore need continuous supervision. Examples of such processes are:

- o Heat treatment
- o Surface treatment
- o Adhesive bonding
- o Composite manufacturing
- o Welding
- o Shot-peening and blasting
- o Forming
- o Chemical milling

These processes are extensively examined and qualified. Special requirements on authorized personnel, work-shops, equipment, environment, etc. exists.

Special routines for items produced in these processes are also carried-out before series-manufacturing is started in order to qualify the process to that specific item. Systematic procedures and documentation of the process is done. The item is checked against design and production instructions, correctness of tools and suitability of process.

For some processes, destructive testing is done on the first item which has been produced in the complete process. This item is cut in parts and tested in laboratory.

A complicated process like the automated process for bonding metal panels includes moments such as handling, cleaning, surface treatment, application of primers and adhesive and curing, and are carefully supervised. Each panel is followed through the complete process by specimens for destructive testing. Each individual panel is also 100% scanned by transmission ultrasonic equipments.

## **6. STRUCTURAL MAINTENANCE PROGRAMME**

### **6.1 Structural maintenance concept**

The Saab 340 structural maintenance programme was developed to satisfy the requirements for damage tolerance that were new at the time for certification. This required the development of new procedures for determining the adequacy of the inspection program for timely detection of damage. The major forms of damage considered during the program development were environmental deterioration (corrosion, stress corrosion), accidental damage and fatigue damage. The results of these analyses gave the initial scheduled maintenance programme for Structural Significant Items (SSI) and zone surveillance. The programme was developed according to the philosophy and logic in the ATA MSG-3 document.

### **6.2 The MSG-3 analysis**

#### **6.2.1 Structural analysis logic**

The structural analysis logic is shown in figure 12 and consists of the following tasks:

- o List per ATA 100 all structure
- o Categorize items as Structural Significant Items "SSI" or as other structure.
- o Document each SSI:
  - Illustrations
  - Function
  - Stress characteristics/level
  - Material & Manufacturing process
  - Protection
- o Classify SSI's as damage tolerant or safe life structure
- o Check if SSI's are fatigue, environmental or accidental critical.
- o Perform fatigue damage analysis for damage tolerant items
- o Select safe life limit for safe life items
- o Perform environmental damage analysis for damage tolerant and safe life items
- o Perform accidental damage analysis for damage tolerant and safe life items
- o Summary of SSI inspection tasks on SSI Task Summary Sheet

The considerations to be taken into account when selecting SSI candidates are given according to the logic flow diagram shown in figure 13.

### **6.2.2 Fatigue damage analysis procedure**

Fatigue critical items are items which are significantly loaded in tension or shear. The fatigue damage analysis procedure follows the philosophy and principles developed in the MSG-3 document. Therefore the tasks necessary to ensure an appropriate fatigue damage detection, are selected based on calculations and ratings of the following main parameters:

- o Critical crack length
- o Threshold associated with the detectable size of fatigue damage
- o Crack growth assessment for repeat inspection intervals
- o Detectable damage size in relation to the various levels of inspection
- o Target values for scheduled maintenance intervals

### **6.2.3 Environmental damage analysis procedure**

Environmental critical items are only those items made of less corrosion resistant materials. The environmental damage analysis procedure follows the philosophy and principles as stated in the ATA MSG-3 document. Necessary tasks to ensure the appropriate detection of environmental damages are selected based on a rating system with the following parameters:

- o Susceptibility to various types of corrosion
- o Type of protective treatments
- o Exposure to adverse type of environment

### **6.2.4 Accidental damage analysis procedure**

Accidental critical items are those items which are likely to be damaged by accidental damage. Manufacturing defects are considered completely random and are supposed to occur on any aircraft at any location. Therefore such defects are taken care of in the damage tolerance analysis and are covered by the fatigue damage analysis items. Accidental damages other than manufacturing defects are also occurring as a random discrete event during operation and maintenance and reduce the inherent level of residual strength. Large size accidental damages caused by engine and propeller disintegration, bird strike and major collision with ground equipment are covered by specific regulations which require evaluation of residual strength capability for short periods of time.

The tasks to ensure detection of accidental damages are determined using a rating system including the following parameters:

- o Kind of damage (Cargo handling, runway debris etc)
- o Likelihood of accidental damage
- o Extent of residual strength.
- o Crack growth rate

### **6.3 The maintenance review board report**

The tasks that are results of the MSG-3 analysis are included in the Maintenance Review Board (MRB) Report, Saab 340 Maintenance Programme, which is the document that outlines the initial maintenance requirements approved by the regulatory authorities.



### **6.3.1 Structural inspection programme.**

Tasks that according to the MSG-3 analyses are defined as directed inspections (detailed or special detailed) are included in "Structural Inspection Programme". Examples of detail inspections in the Saab 340 structural inspection programme are shown in figure 14.

For Saab 340, the threshold for first inspection is in general 16000 flights. Other thresholds are 28000 and 40000 flights. Repeat intervals are 3000, 6000 and 12000 flights.

### **6.3.2 Zonal inspection programme.**

Inspection tasks for which general visual inspections, were judged to be appropriate are transferred to the Zonal Inspection Programme.

Threshold and repeat intervals for environmental damage inspection in general are 6 and 4 years respectively. Accidental damage inspection, in general, refers to LC, A, B and C check.

## **7. MANUALS AND CUSTOMER SUPPORT**

### **7.1 Structural repair manual**

A structural repair manual has been prepared in accordance with ATA 100 specification to include descriptive information and specific instructions and data relative to the field repair of the structure of the Saab 340 aircraft.

It contains structural material identification, allowable structural damage criteria and repair designs applicable to structural components of the aircraft that are most likely to be damaged. In addition, it contains procedures to be performed during structural repair such as fastener installation, protective treatment and sealing of repair parts and other processes.

Based on field experience of fatigue, environmental and accidental damages the manual is continuously updated with new repair schemes

### **7.2 Corrosion prevention manual**

In order to comply with the need to set up corrosion control programmes to ageing aircraft, a corrosion prevention manual is being prepared. The manual will give complete coverage for:

- o Identification and type of corrosion
- o Corrosion prevention
- o Inspection and detection
- o Corrosion removal techniques

The manual is planned to be released during the second half of 1992.

### **7.3 Customer support**

For special repairs not covered by the structural repair manual, the operators of the Saab 340 are assisted 24 hours a day by a Saab Product Support Team with experience from stress engineering as well as practical structural repair. The team is also backed up by Saab Aircraft Division Stress Analysis & Testing, Airframe Design & Engineering Technology and Materials & Process Technology departments.

Saab personnel are also on site for assistance and training when the first structural inspection, at 16,000 flights, is made for the first time for each operator.

## **8. FEEDBACK FROM OPERATORS**

### **8.1 The age exploration programme - AEP**

This programme is used to systematically evaluate inspection tasks based on analysis of collected information from in-service experience. The programme can be performed by individual operators or by groups of smaller operators.

It is used for environmental deterioration surveillance and to assess an items resistance to environmental deterioration with respect to increasing age. Inspection of the oldest aircraft of the fleet will give the most rapid data from in-service experience and should lead to minimizing of the total number of required inspections.

AEP will only apply to environmental damage having a systematical characteristic. This means that damages occurring on a random basis caused by stress corrosion, corrosion from fluid spillage and other accidental damages can not be evaluated by this program. AEP will only apply when directed inspections are judged as necessary. If no damages are found only zonal inspection will be needed for those aircraft not subject to this program.

### **8.2 Reporting of structural damages**

All significant structural discrepancies will be reported by the procedures already established by the airworthiness authorities e.g. Service Difficulty Reports.

Each operator is requested according to the MRB document to report to Saab discrepancies found during scheduled inspections. Where practical, details of the damages found and the inspection methods used should be reported as well as any other damage found by subsequent examination.

### **8.3 Special investigation of the operational usage.**

The fatigue and damage tolerance sizing of Saab 340 was based on a usage defined already in the project phase, early 1980. The missions, which were considered to be conservative at that time, consist of flights distributed on four cruise altitudes having an average flight time of 30 minutes.

It is supposed that the difference in usage between individual operators might be significant for commuter aircraft. Therefore, in order to validate the fatigue spectra for Saab 340A, an inquiry about the operational usage among the operators was performed during Nov. 1988 to Nov. 1989. By assistance of Saab Product Support, the following information was requested:

- o Routes and corresponding distance, flight time, cruise altitude, no. of flights per week.
- o Weight and CG:load sheets (weight and balance) from representative airports.
- o Power application at take off and use of reverse at landing.
- o Settings for take off flap, approach flap and landing flap.
- o Speeds for retraction of take off flap and for extension of approach flap, landing flap and landing gear.
- o Speeds for climb, cruise and descent.



Information was received from more than 80% of the operators, thus forming a good base for the statistics.

The investigation revealed a larger portion of flights at high altitudes, and consequently a more severe cabin pressurization spectrum, than anticipated during the design phase of the aircraft.

A fatigue investigation was done which resulted in an update of the cabin pressurization design spectrum. This led also to an update of the cabin pressurization of the full-scale fatigue test at 120,000 simulated flights. Additional pressurization cycles were applied before the restart with the new spectrum in order to simulate a more severe loading from the beginning of the test.

A review of all damage tolerance analysis which was affected of this change in spectrum was also done. The damage tolerance analysis for Saab 340B was done from the beginning with the new spectrum.

Referring to the figure 4, this is an example of how the feedback from field experiences is integrated in the fatigue and damage tolerance sizing concept resulting in a preserved control of the structural integrity.

#### **ACKNOWLEDGEMENTS**

This paper presents an approach, which comprises different technologies at Saab Aircraft and is a result of a joint effort. Those who have contributed to this paper are gratefully acknowledged. Major contributions were prepared by Mr Göte Strindberg, Engineering Technology, Mr Einar Hultgren, Materials and process technology, Mr Göran Prestby, Stress Analysis Commercial Aircraft, Mr Lars Sjöström, Loads and structural analysis technology, Mr Göran Lindström and Mr Håkan Persson, Saab Product Support.

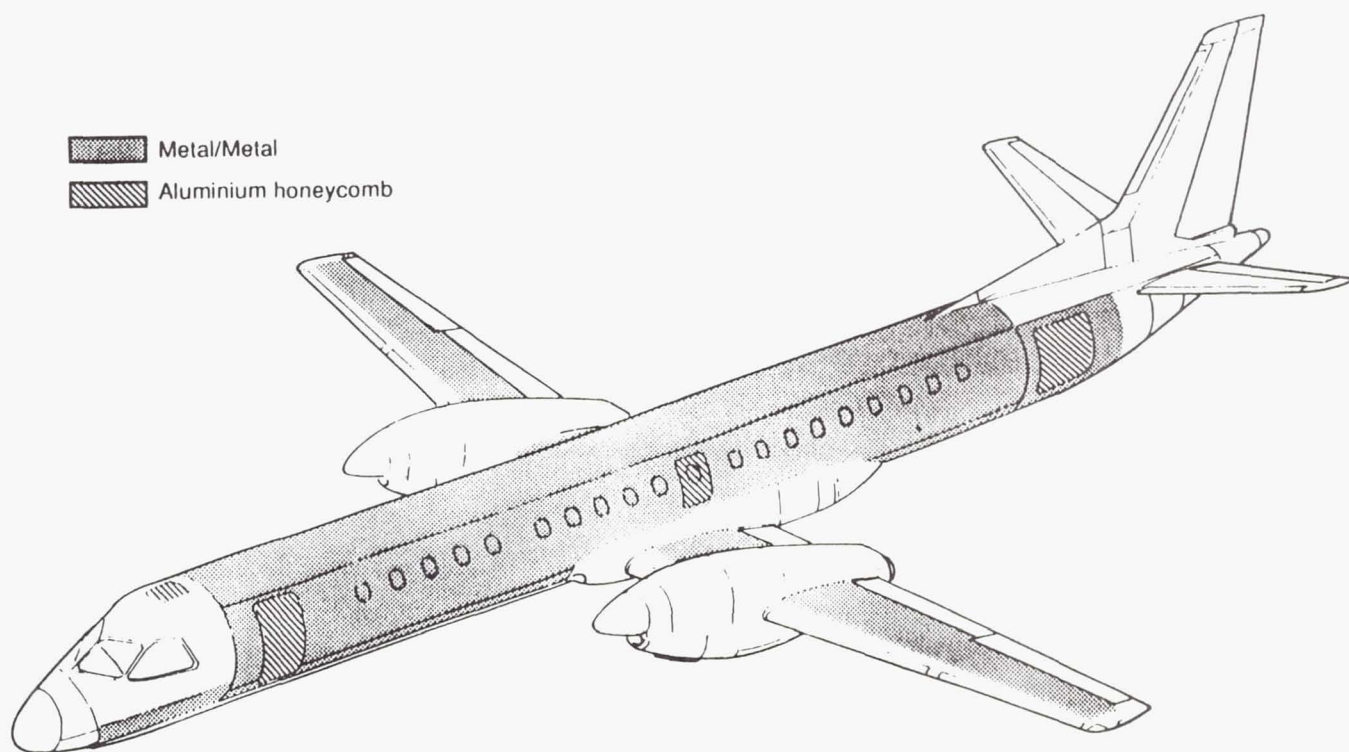


Figure 1. The metal to metal bonded structure on Saab 340

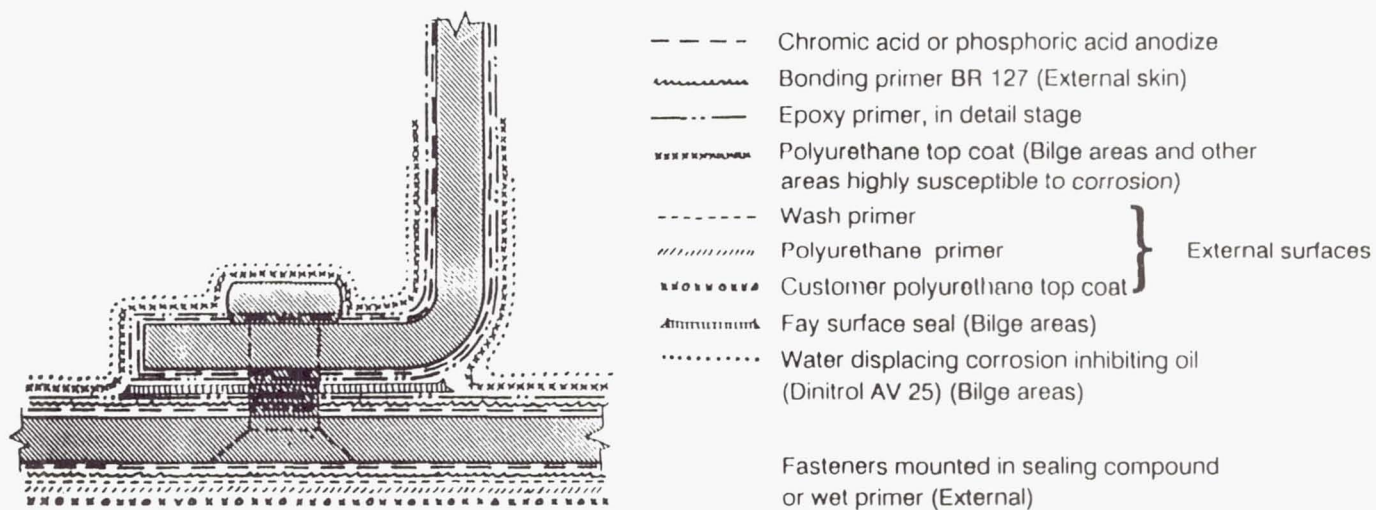


Figure 2. The corrosion protection scheme

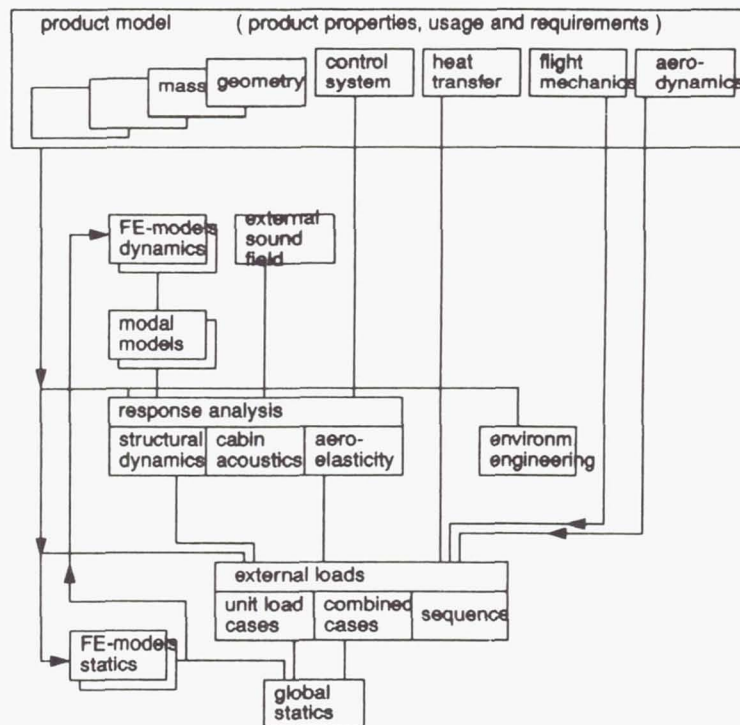


Figure 3. The current basic process for loads analysis

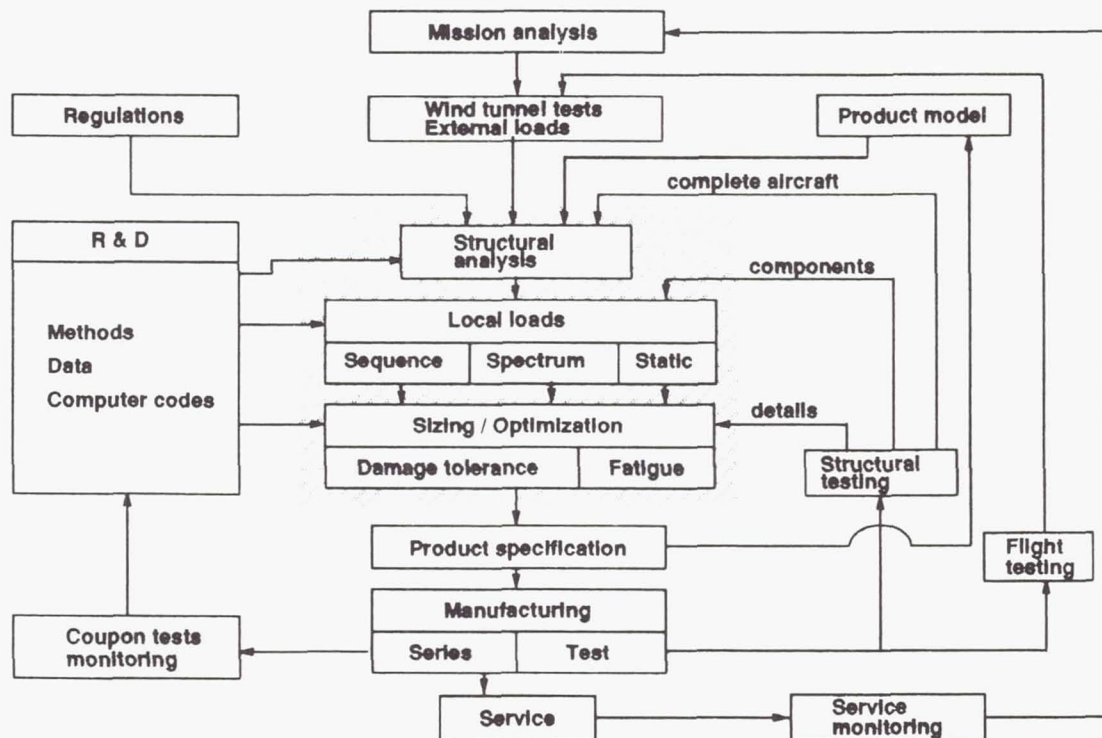


Figure 4. The sizing and verification concept





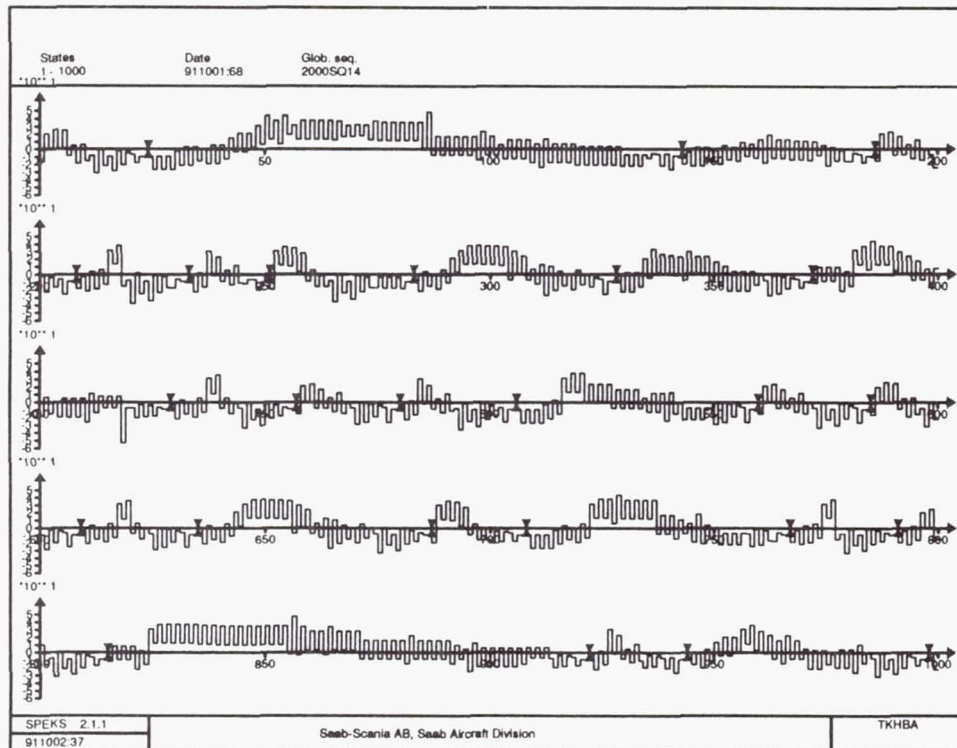


Figure 7. A part of the reduced local axial stress sequence of the inner flange.

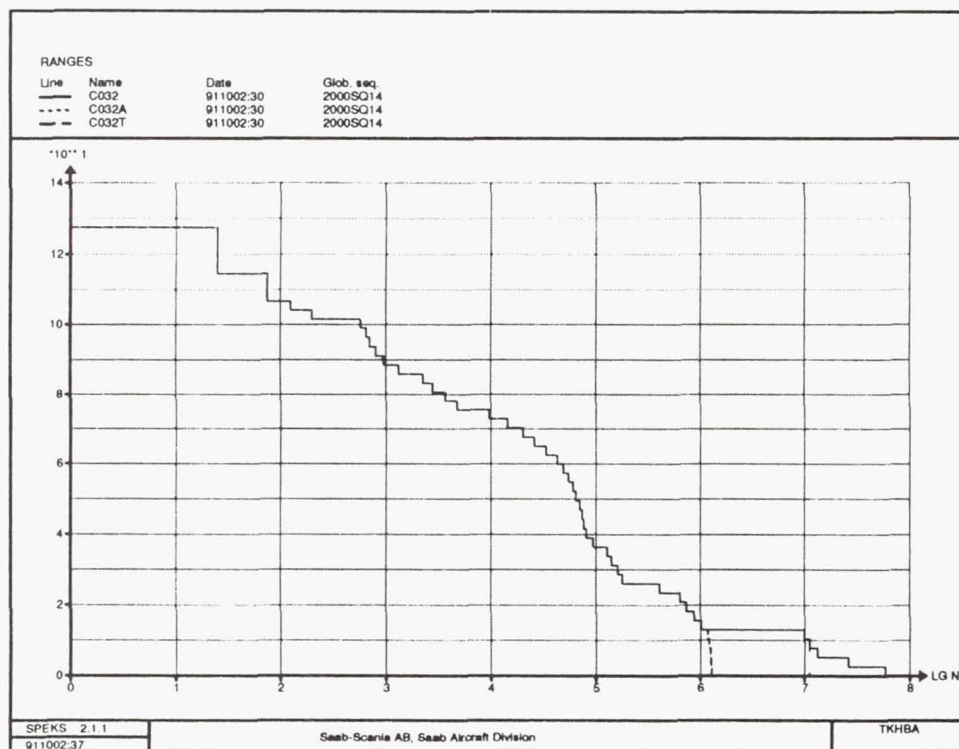


Figure 8. The distribution of rain-flow counted ranges for the original and reduced sequence.

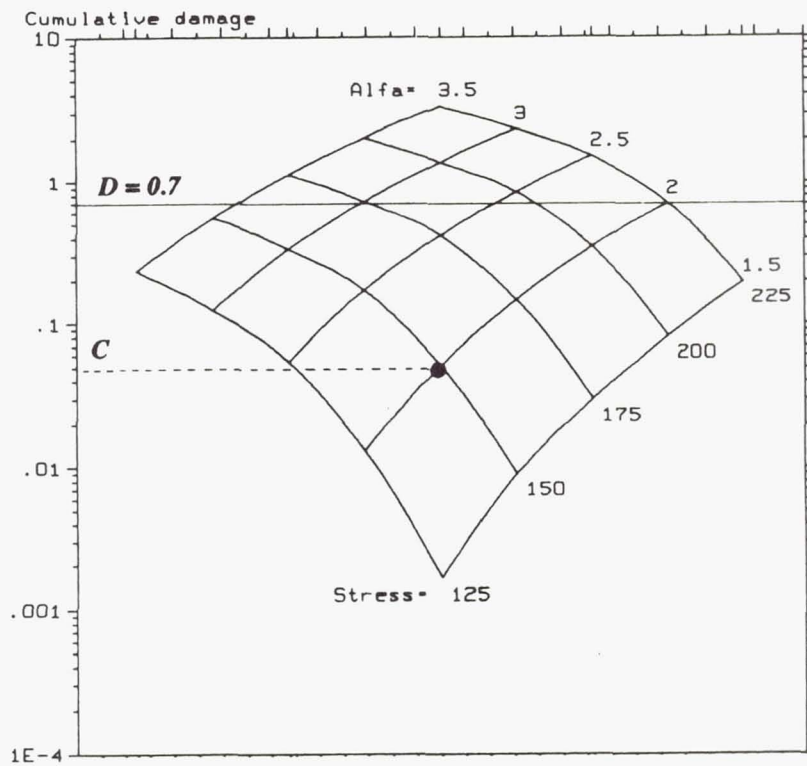


Figure 9. A plot of cumulative damage sums

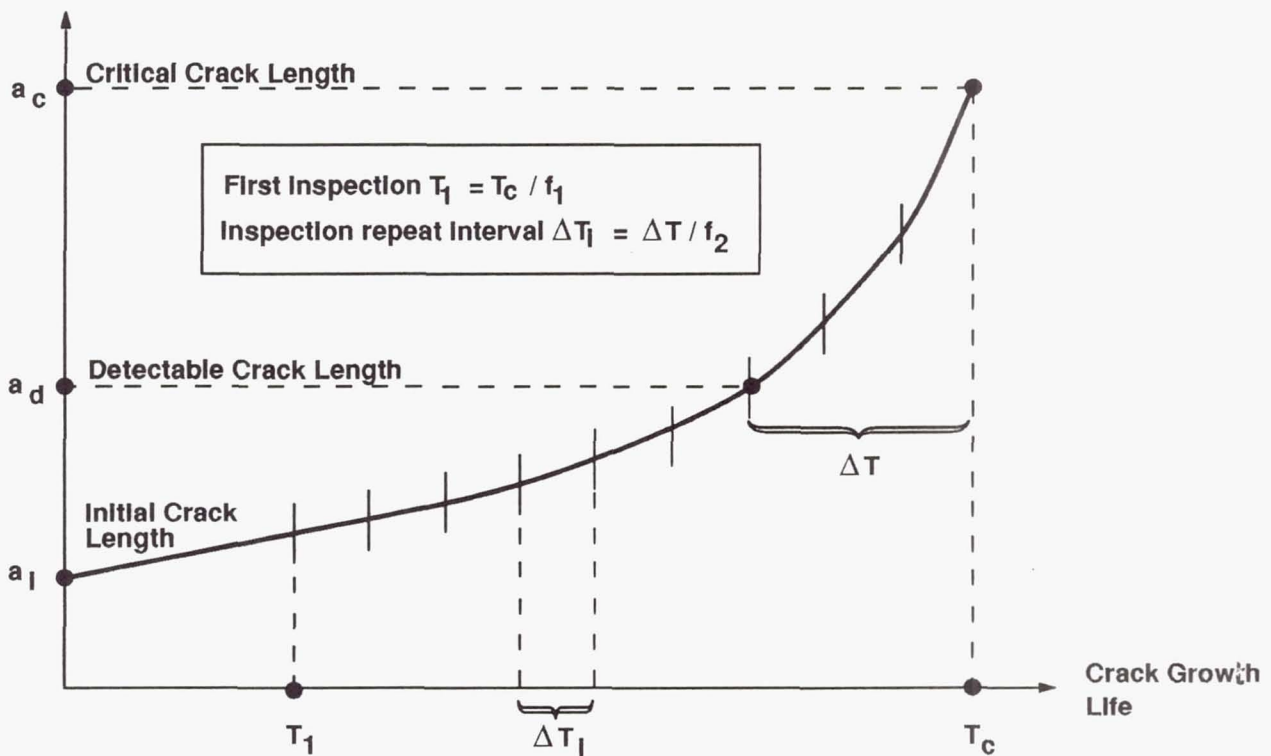
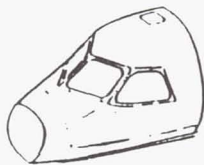


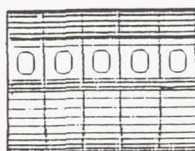
Figure 10. The crack growth curve with threshold and inspection repeat intervals



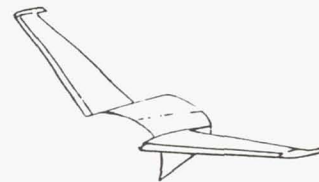
FORWARD FUSELAGE PRESSURE TEST



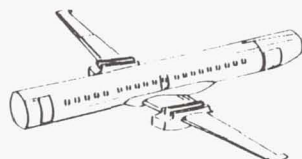
CABIN PANEL FATIGUE AND DAMAGE TOLERANCE TEST



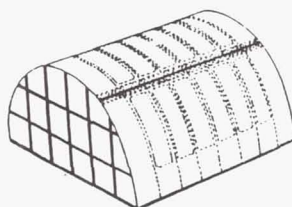
HORIZONTAL STABILIZER FATIGUE TEST



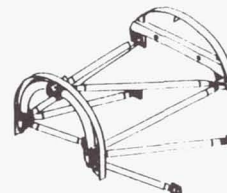
WING/FUSELAGE FATIGUE TEST



REAR FUSELAGE PANEL DAMAGE TOLERANCE TEST



ENGINE MOUNTING STRUCTURE FATIGUE TEST



FLAP FATIGUE TEST

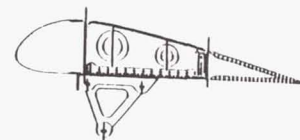


Figure 11. Major fatigue and damage tolerance tests on Saab 2000

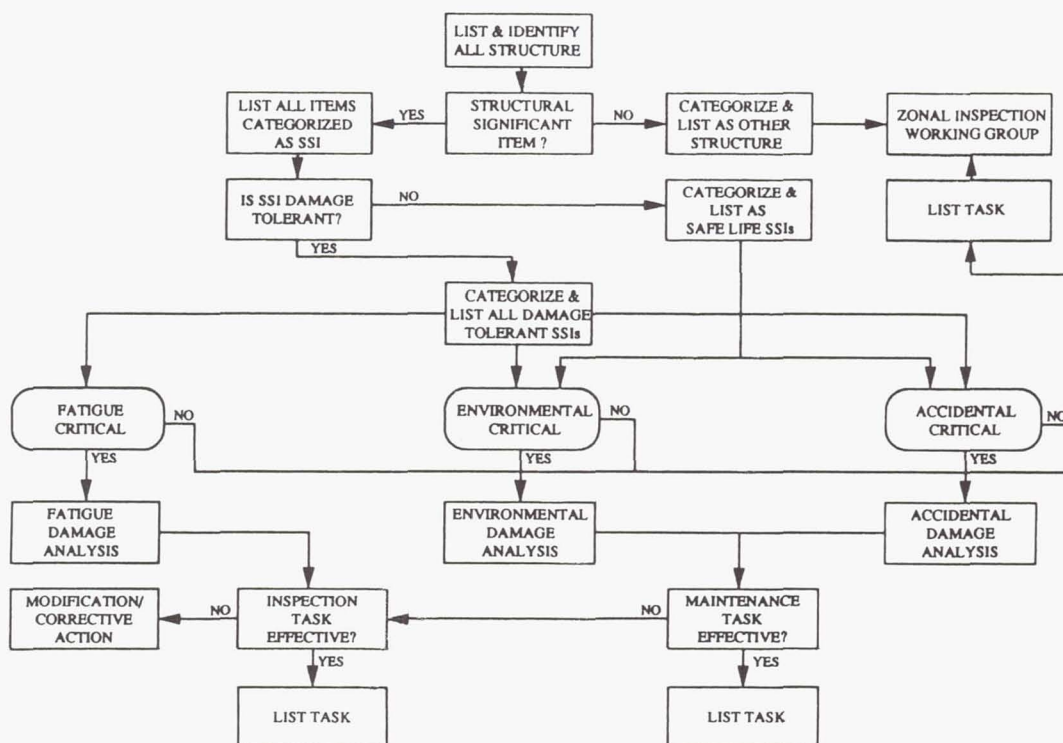


Figure 12. The MSG-3 structural analysis logic

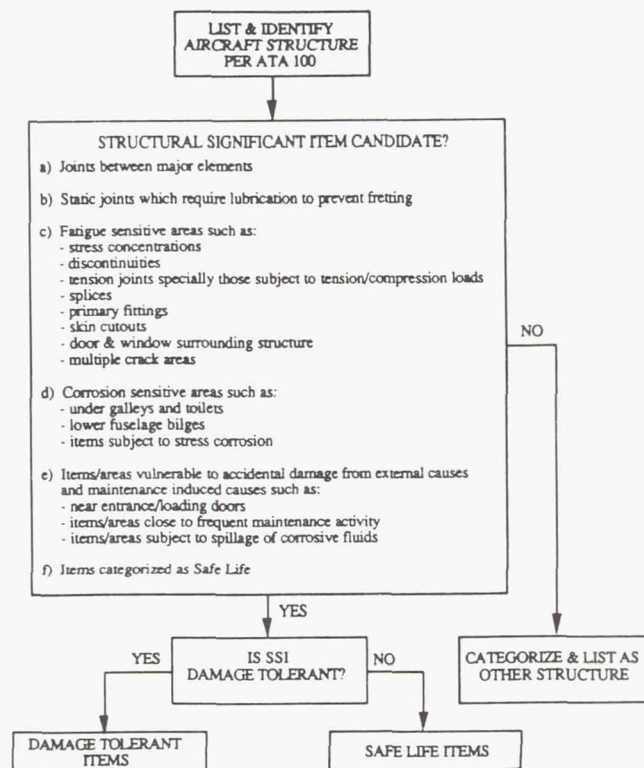


Figure 13. The logic flow diagram for selecting SSI candidates.

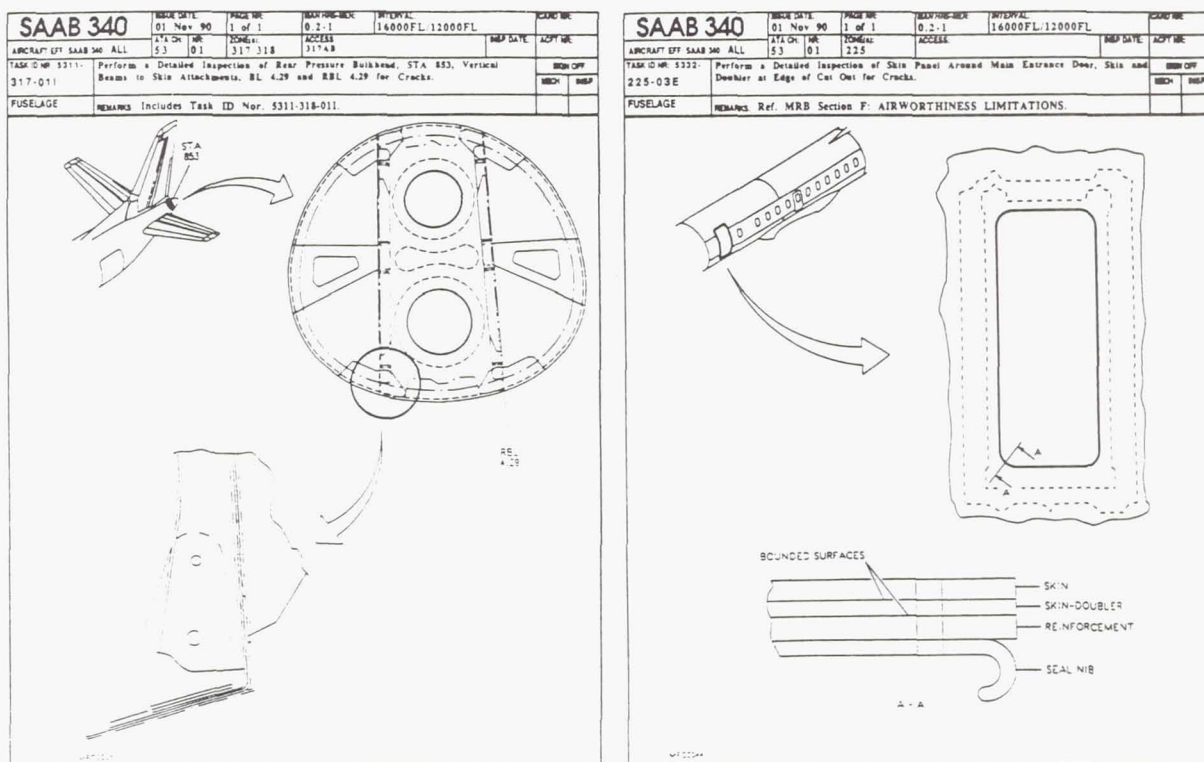


Figure 14. Examples of detail inspections.

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# SIMPLIFIED COMPUTATIONAL METHODS FOR ELASTIC AND ELASTIC-PLASTIC FRACTURE PROBLEMS

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## SUMMARY

This paper presents an overview of some of the recent (1984-1991) developments in computational/analytical methods in the mechanics of fracture: (i) analytical solutions for elliptical or circular cracks embedded in isotropic or transversely isotropic solids (the crack-plane being at an arbitrary angle to the axis of transverse isotropy), with crack-faces being subjected to arbitrary tractions; (ii) finite-element or boundary-element alternating methods for two- and three-dimensional crack problems; (iii) a "direct-stiffness" method for stiffened panels with flexible fasteners and with multiple cracks, using the alternating method; (iv) multiple-site-damage near a row of fastener holes; (v) analysis of cracks with bonded repair patches; (vi) methods for generation of weight-functions for 2- and 3-D crack problems, and (vii) domain-integral methods for elastic-plastic or inelastic crack mechanics.

## INTRODUCTION

The starting point for this overview is the monograph on "Computational Methods in the Mechanics of Fracture" [1], with contributions by several noted researchers. The various articles in that monograph were prepared by individual authors in the 1984-85 time frame, with the material being current mostly as of 1984. In the present paper, some recent advances in computational fracture mechanics in the intervening years are summarized. The coverage of topics is limited to those listed in the "Summary" above; and furthermore the scope of the article is limited by the authors' own interests.

In each of the topics listed in the "Summary", a reasonably self-contained account of the generic issues, and the progress made in addressing them, is provided below.

Elastic and elastic-plastic fracture mechanics methodologies, that are easy to apply, are mandatory in a strategy for assessing the integrity of cracked structures. Straight-forward finite element/boundary-element analyses of such problems, with detailed numerical modeling of crack-tip singularities, would be prohibitively expensive, even in the linear elastic regime. Furthermore, such straight-forward numerical analyses are useful for a single configuration of crack(s) and loads; and whenever the crack-configuration or the load-configuration changes, the entire analysis must be repeated. Thus, alternate strategies for efficient analyses of structural integrity in the presence of multiple-site-damage are needed. This paper reviews some such possible strategies.

## 1 ANALYTICAL SOLUTIONS FOR ELLIPTICAL OR CIRCULAR CRACKS IN ISOTROPIC OR TRANSVERSELY ISOTROPIC SOLIDS, WITH ARBITRARY CRACK-FACE TRACTIONS

In practice, the actual flaws in three-dimensional structural components are often approximated by elliptical cracks. For this reason, the problem of an embedded elliptical crack in an infinite solid has been the focus of a considerable number of studies.

The authors in [2] have presented a general solution procedure for an embedded elliptical crack in an isotropic infinite solid, subject to arbitrary crack-face tractions. Later, the authors in [3] have refined

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<sup>1</sup>Institute Professor and Director

and completed this solution, deriving (i) alternative non-singular forms for linear algebraic equations relating crack-face tractions and potential functions, (ii) a general procedure for the evaluation of the required elliptic integrals, and (iii) a systematic procedure for the evaluation of the partial derivatives of the potential functions. These derivations made it possible to extract the analytical close-form solution for any polynomial order of crack-face tractions. For later convenience, we cite this general solution altogether [2] and [3] as VNA solution. The VNA solution represents a generalization, of the potential representation of [4] and [5].

In the following, we present a brief summary of the VNA solution. Further details of the VNA solution can be found in the cited original papers.

### 1.1 THE VNA SOLUTION (AN ELLIPTICAL CRACK IN AN ISOTROPIC SOLID)

Suppose that  $x_1$  and  $x_2$  are Cartesian coordinates in the plane of the elliptical crack and  $x_3$  is normal to the crack-plane such that:

$$(x_1/a_1)^2 + (x_2/a_2)^2 = 1, \quad a_1 > a_2 \quad (1)$$

describes the border of the elliptical crack of aspect ratio  $(a_1/a_2)$ . The necessary ellipsoidal coordinates  $\xi_\alpha$  ( $\alpha = 1, 2, 3$ ) are defined as the roots of the cubic equation

$$\omega(s) = 1 - \left( \frac{x_1^2}{a_1^2 + s} \right) - \left( \frac{x_2^2}{a_2^2 + s} \right) - \left( \frac{x_3^2}{s} \right) = 0 \quad (2)$$

where

$$\infty > \xi_3 \geq 0 \geq \xi_2 \geq -a_2^2 \geq \xi_1 \geq -a_1^2$$

so that the interior of the ellipse is given by  $\xi_3 = 0$  and its boundary by  $\xi_2 = \xi_3 = 0$ .

Let the tractions along the crack-surface be expressed in the form

$$\sigma_{3\alpha}^{(0)} = \sum_{i=0}^1 \sum_{j=0}^1 \sum_{m=0}^M \sum_{n=0}^m A_{\alpha, m-n, n}^{(i, j)} x_1^{2m-2n+i} x_2^{2n+j}; \quad (\alpha = 1, 2, 3) \quad (3)$$

so that the values of  $(i, j)$  specify the symmetries of the load with respect to the axes of the ellipse.  $M$  is an arbitrary integer which is related to the order of the polynomial. The solution corresponding to the load expressed by Eq. (3) can be assumed in terms of the potential functions

$$f_\alpha = \sum_{i=0}^1 \sum_{j=0}^1 \sum_{k=0}^M \sum_{\ell=0}^k C_{\alpha, k-\ell, \ell}^{(i, j)} F_{2k-2\ell+i, 2\ell+j}; \quad (\alpha = 1, 2, 3) \quad (4)$$

where

$$F_{2k-2\ell+i, 2\ell+j} = \frac{\partial^{2k+i+j}}{\partial x_1^{2k-2\ell+i} \partial x_2^{2\ell+j}} \int_{\xi_3}^{\infty} [\omega(s)]^{2k+i+j+1} \frac{ds}{\sqrt{Q(s)}} \quad (5)$$

and  $Q(s) = s(s+a_1^2)(s+a_2^2)$ . The components of displacement  $u_i$  and stress  $\sigma_{ij}$  in terms of  $f_\alpha$  ( $\alpha = 1, 2, 3$ ) are given by

$$\begin{aligned} u_1 &= (1-2\nu)(f_{1,3} + f_{3,1}) - (3-4\nu)f_{1,3} + x_3(\nabla \cdot \hat{\mathbf{f}})_{,1} \\ u_2 &= (1-2\nu)(f_{2,3} + f_{3,2}) - (3-4\nu)f_{2,3} + x_3(\nabla \cdot \hat{\mathbf{f}})_{,2} \\ u_3 &= -(1-2\nu)(f_{1,1} + f_{2,2}) - 2(1-\nu)f_{3,3} + x_3(\nabla \cdot \hat{\mathbf{f}})_{,3} \end{aligned} \quad (6)$$

and

$$\begin{aligned}
\sigma_{11} &= 2\mu[f_{3,11} + 2\nu f_{3,22} - 2f_{1,31} - 2\nu f_{2,32} + x_3(\nabla \cdot \hat{\mathbf{f}})_{,11}] \\
\sigma_{22} &= 2\mu[f_{3,22} + 2\nu f_{3,11} - 2f_{2,32} - 2\nu f_{1,31} + x_3(\nabla \cdot \hat{\mathbf{f}})_{,22}] \\
\sigma_{12} &= 2\mu[(1 - 2\nu)f_{3,12} - (1 - \nu)(f_{1,23} + f_{2,13}) + x_3(\nabla \cdot \hat{\mathbf{f}})_{,12}] \\
\sigma_{33} &= 2\mu[-f_{3,33} + x_3(\nabla \cdot \hat{\mathbf{f}})_{,33}] \\
\sigma_{31} &= 2\mu[-(1 - \nu)f_{1,33} + \nu(f_{1,11} + f_{2,21}) + x_3(\nabla \cdot \hat{\mathbf{f}})_{,13}] \\
\sigma_{32} &= 2\mu[-(1 - \nu)f_{2,33} + \nu(f_{1,12} + f_{2,22}) + x_3(\nabla \cdot \hat{\mathbf{f}})_{,23}]
\end{aligned} \tag{7}$$

and

$$\nabla \cdot \hat{\mathbf{f}} = f_{1,1} + f_{2,2} + f_{3,3} \tag{8}$$

where  $\mu$  and  $\nu$  are the shear modulus and Poisson's ratio.

By successive differentiation, it can be shown from (5) that, since  $\omega(\xi_3) = 0$ ,

$$F_{k\ell} = \int_{\xi_3}^{\infty} \frac{\partial^{k+\ell} \omega^{k+\ell+1}}{\partial x_1^k \partial x_2^\ell} \frac{ds}{[Q(s)]^{1/2}} \equiv \int_{\xi_3}^{\infty} \partial_1^{k_1} \partial_2^{\ell_1} \partial_3^{m_1} \omega^{k+\ell+1} \frac{ds}{[Q(s)]^{1/2}} \tag{9}$$

wherein  $(2k - 2\ell + i)$  and  $(2\ell + j)$  in Eq. (24) are replaced by  $k$  and  $\ell$  in the above equation and

$$k_1 = k, \quad \ell_1 = \ell, \quad m_1 = 0$$

In (9) we have used the additional notation that  $\partial_\alpha^j$  implies the  $j$ th partial derivative with respect to  $x_\alpha$ . Similarly, the first-order partial derivatives of  $F_{k\ell}$  with respect to  $x_\beta$  ( $\beta = 1, 2, 3$ ) can be expressed by

$$F_{k\ell,\beta} = \int_{\xi_3}^{\infty} \partial_1^{k_1} \partial_2^{\ell_1} \partial_3^{m_1} \omega^{k+\ell+1} \frac{ds}{[Q(s)]^{1/2}} \tag{10}$$

where

$$k_1 = k + \delta_{1\beta}, \quad \ell_1 = \ell + \delta_{2\beta}, \quad m_1 = \delta_{3\beta}$$

and  $\delta_{1\beta}$ , etc., are the well-known Kronecker deltas.

In the case of the second- and third-order partial derivatives, we derive:

$$F_{k\ell,\beta\gamma} = \int_{\xi_3}^{\infty} \partial_1^{k_1} \partial_2^{\ell_1} \partial_3^{m_1} \omega^{k+\ell+1} \frac{ds}{[Q(s)]^{1/2}} + F_{k\ell\beta\gamma}^0 \tag{11}$$

where

$$F_{k\ell\beta\gamma}^0 = (k + \ell + 1)! \frac{x_1^{k_1} x_2^{\ell_1} x_3^{m_1}}{(\xi_3 - \xi_1)(\xi_3 - \xi_2)} \{\rho_1^{k_1} \rho_2^{\ell_1} \rho_3^{m_1} [Q(s)]^{1/2}\}_{s=\xi_3} \tag{12}$$

$$k_1 = k + \delta_{1\beta} + \delta_{1\gamma}, \quad \ell_1 = \ell + \delta_{2\beta} + \delta_{2\gamma}, \quad m_1 = \delta_{3\beta} + \delta_{3\gamma} \tag{13}$$

$$\rho_\alpha = \frac{\partial^2 \omega}{\partial x_\alpha^2} = -2/(a_\alpha^2 + s), \quad (\alpha = 1, 2, 3) \tag{14}$$

in which  $a_3 \equiv 0$ , and

$$F_{k\ell,\beta\gamma\delta} = \int_{\xi_3}^{\infty} \partial_1^{k_1} \partial_2^{\ell_1} \partial_3^{m_1} \omega^{k+\ell+1} \frac{ds}{[Q(s)]^{1/2}} + \frac{\partial F_{k\ell\beta\gamma}^0}{\partial x_\delta} + G^0 \tag{15}$$



where

$$\begin{aligned}
G^0 &= \frac{(k + \ell + 1)! [Q(s)]^{1/2}}{(\xi_3 - \xi_1)(\xi_3 - \xi_2)} \rho_1^{k_1} \rho_2^{\ell_1} \rho_3^{m_1} x_1^{k_1} x_2^{\ell_1} x_3^{m_1} \\
&\quad \times \left[ \frac{k_0(k_0 - 1)}{2\rho_1 x_1^2} + \frac{\ell_0(\ell_0 - 1)}{2\rho_2 x_2^2} + \frac{m_0(m_0 - 1)}{2\rho_3 x_3^2} \right]_{s=\xi_3} \\
k_0 &= k + \delta_{1\beta} + \delta_{1\gamma}; \quad \ell_0 = \ell + \delta_{2\beta} + \delta_{2\gamma}; \quad m_0 = \delta_{3\beta} + \delta_{3\gamma} \\
k_1 &= k_0 + \delta_{1\delta}; \quad \ell_1 = \ell_0 + \delta_{2\delta}; \quad m_1 = m_0 + \delta_{3\delta}
\end{aligned} \tag{16}$$

A systematic procedure for the evaluation of the partial derivatives of  $F_{k\ell\beta\alpha}^0$  in (15) is given in [3]. It is noted that these derivatives are needed (i) in satisfying the boundary conditions on the crack-face and (ii) in evaluating the far-field stresses in the solid containing the elliptical crack which is subject to arbitrary tractions.

It is now seen from (9)–(15) that one needs to evaluate a generic integral of the type:

$$\int_{\xi_3}^{\infty} \partial_1^{k_1} \partial_2^{\ell_1} \partial_3^{m_1} \omega^{k+\ell+1} \frac{ds}{[Q(s)]^{1/2}} \tag{17}$$

To accomplish this, we expand  $\omega^{k+\ell+1}$  in terms of  $x_\alpha^2$  and carry out the indicated differentiations term by term. Thus, one obtains:

$$\begin{aligned}
\int_{\xi_3}^{\infty} \partial_1^{k_1} \partial_2^{\ell_1} \partial_3^{m_1} \omega^{k+\ell+1} \frac{ds}{[Q(s)]^{1/2}} &= (k + \ell + 1)! \sum_{p=0}^{k+\ell+1} \sum_{q=0}^p \sum_{r=0}^q \frac{(-1)^p}{(k + \ell + 1 - p)!} \\
&\quad \times \frac{(2p - 2q)! (2p - 2r)! (2r)!}{(p - q)! (q - r)! (r)!} \frac{x_1^{2p-2q-k_1}}{(2p - 2q - k_1)!} \\
&\quad \times \frac{x_2^{2q-2r-\ell_1}}{(2q - 2r - \ell_1)!} \frac{x_3^{2r-m_1}}{(2r - m_1)!} J_{p-q,q-r,r}(\xi_3)
\end{aligned} \tag{18}$$

where

$$J_{p-q,q-r,r}(\xi_3) = \int_{\xi_3}^{\infty} \frac{ds}{(s + a_1^2)^{p-q} (s + a_2^2)^{q-r} (s + a_3^2)^r [Q(s)]^{1/2}} \tag{19}$$

In general, the integral indicated in Eq. (19), for a given set of parameters  $p, q, r$ , can be evaluated in terms of incomplete elliptic integrals of the first and second kinds, and Jacobian elliptic functions. The derivation of the closed-form expressions involves exorbitant work even for relatively lower-order components of  $J_{p-q,q-r,r}$  (see [6]). Therefore, the derivation of a systematic generic procedure for the evaluation of the elliptic integrals  $J_{p-q,q-r,r}$  was important in the development as well as in the numerical implementation of the VNA solution. A procedure for this has been developed in [3] and is summarized as follows.

Eq. (19) can be rewritten in terms of Jacobian elliptic functions, as:

$$\begin{aligned}
J_{p-q,q-r,r} &= \frac{2}{a_1^{2p+1}} \int_0^{u_1} (\text{sn}^{2p} u) (\text{nd}^{2q-2r} u) (\text{nc}^{2r} u) du \\
&\equiv \frac{2}{a_1^{2p+1}} L_{p,q-r,r}
\end{aligned} \tag{20}$$

where

$$\text{sn}^2 u_1 = a_1^2 / (a_1^2 + \xi_3)$$

The following identities for Jacobian elliptic functions are used:

$$\begin{aligned} \operatorname{sn}^2 u_1 + \operatorname{cn}^2 u &= 1; & k^2 \operatorname{sn}^2 u + \operatorname{dn}^2 u &= 1 \\ \operatorname{dn}^2 u - k^2 \operatorname{cn}^2 u &= k'^2; & k'^2 \operatorname{sn}^2 u + \operatorname{cn}^2 u &= \operatorname{dn}^2 u \\ \operatorname{tnu} &= \operatorname{sn} u / \operatorname{cn} u; & \operatorname{dcu} &= \operatorname{dn} u / \operatorname{cn} u; & \operatorname{cd} u &= \operatorname{cn} u / \operatorname{dn} u \\ \operatorname{nd} u &= 1 / \operatorname{dn} u; & \operatorname{nc} u &= 1 / \operatorname{cn} u; & \operatorname{sd} u &= \operatorname{sn} u / \operatorname{dn} u \end{aligned} \quad (21)$$

where

$$k^2 = (a_1^2 - a_2^2) / a_1^2; \quad k'^2 = 1 - k^2 \quad (22)$$

By using integration by parts in (20), one sees that:

$$\begin{aligned} L_{p,q-r,r} &= \frac{1}{(2r-1)k'^2} \{ (\operatorname{sn}^{2p+1} u) (\operatorname{nc}^{2r-1} u) (\operatorname{nd}^{2q-2r-1} u) \big|_0^{u_1} \\ &\quad + [2(-p+r-1) + 2(p-q-r+2)k^2] L_{p,q-r,r-1} \\ &\quad + k^2(-2p+2q-3) L_{p,q-r,r-2} \} \end{aligned} \quad (23)$$

Thus, one needs the starting values of  $L_{p,q-r,r-1}$  and  $L_{p,q-r,r-2}$  to evaluate  $L_{p,q-r,r}$ . The lowest-order starting values are:

$$\begin{aligned} L_{p,q,-1} &= \int_0^{u_1} \operatorname{sn}^{2p} u \operatorname{nd}^{2q} u \operatorname{nc}^{-2} u \, du \\ L_{p,q,-2} &= \int_0^{u_1} \operatorname{sn}^{2p} u \operatorname{nd}^{2q} u \operatorname{nc}^{-4} u \, du \end{aligned} \quad (24)$$

The above integrals can be reduced to the forms:

$$\begin{aligned} L_{p,q,-1} &= \frac{1}{k^{2p+2}} \sum_{j=0}^p \sum_{\gamma=0}^1 \frac{(-1)^{j+\gamma+1} k'^{2(1-\gamma)} p!}{(p-j)! j! (1-\gamma)! \gamma!} I_{2(q-j-\gamma)} \\ L_{p,q,-2} &= \frac{1}{k^{2p+4}} \sum_{j=0}^p \sum_{\gamma=0}^2 \frac{(-1)^{j+\gamma+2} k'^{2(2-\gamma)} p! 2}{(p-j)! j! (2-\gamma)! \gamma!} I_{2(q-j-\gamma)} \end{aligned} \quad (25)$$

where

$$I_{2m} = \int_0^{u_1} \operatorname{nd}^m u \, du \quad (26)$$

$$I_{2m+2} = \frac{2m(2-k^2)I_{2m} + (1-2m)I_{2m-2} - k^2 \operatorname{sn} u_1 \operatorname{cn} u_1 \operatorname{nd}^{2m+1} u_1}{(2m+1)k'^2} \quad (27)$$

For  $2(p-j-\gamma) < 0$  in (44), we find  $I_{-2m} = G_{2m}$ , where

$$G_{2m+2} = \frac{k^2 \operatorname{dn}^{2m-1} u_1 \operatorname{sn} u_1 \operatorname{cn} u_1 + (1-2m)k'^2 G_{2m-2} + 2m(2-k^2)G_{2m}}{(2m+1)} \quad (28)$$

Thus, finally we see that one needs the following starting values for evaluating the general terms of  $I_{2m+2}$  and  $G_{2m+2}$ :

$$\begin{aligned} I_0 &= G_0 = F(u_1) = u_1 \\ I_2 &= (1/k'^2)[E(u_1) - k^2 \operatorname{sn} u_1 \operatorname{cd} u_1] \\ G_2 &= E(u_1) \end{aligned} \quad (29)$$

where  $F(u_1)$  and  $E(u_1)$  are incomplete elliptic integrals of the first and second kinds, respectively.

Now, the boundary conditions on the crack-face ( $\sigma_{3\alpha} = \sigma_{3\alpha}^0$ ) can be expressed in terms of the potential functions, as follows:

$$\begin{aligned}\sigma_{33}^{(0)} &= -2\mu f_{3,33} \\ \sigma_{3\alpha}^{(0)} &= -2\mu[(1-\nu)f_{\alpha,33} - \nu(f_{1,1\alpha} + f_{2,2\alpha})]\alpha = 1, 2\end{aligned}\quad (30)$$

in which the boundary condition for  $f_3$  is uncoupled from  $f_1$  and  $f_2$ . However, if Eqs. (30) are used directly as in [2], finite parts of the singular terms in the equations relating the coefficients  $C$  of (4) to coefficients  $A$  of Eq. (3) have to be considered. Alternative non-singular forms for the boundary conditions may also be used. Since  $f_\alpha$  ( $\alpha = 1, 2, 3$ ) are harmonic functions, it is seen that:

$$f_{\alpha,33} = -f_{\alpha,11} - f_{\alpha,22} \quad (\alpha = 1, 2, 3) \quad (31)$$

Then, (30a, b) can be rewritten as follows:

$$\begin{aligned}\sigma_{33}^{(0)} &= 2\mu(f_{3,11} + f_{3,22}) \\ \sigma_{3\alpha}^{(0)} &= 2\mu[(1-\nu)(f_{\alpha,11} + f_{\alpha,22}) + \nu(f_{1,1\alpha} + f_{2,2\alpha})]\end{aligned}\quad (32)$$

Substituting (3) and (4) into (32a, b), we obtain the following linear algebraic equations, upon comparing coefficients of like powers in the polynomial series. The relation between the parameters  $A$  and parameters  $C$  can be summarized in a matrix form:

$$\begin{matrix} \{A\} \\ N \times 1 \end{matrix} = \begin{matrix} [B] \\ N \times N \end{matrix} \begin{matrix} \{C\} \\ N \times 1 \end{matrix} \quad (33)$$

where  $N$  is the total number of coefficients  $A$  or  $C$ . For a complete polynomial expressed by (3), the maximum degree of the polynomial  $M_c$  and the number of coefficients  $N$  can be expressed, respectively, as  $M_c = 2M + 1$  and  $N = (M + 1)(2M + 3) \times 3$ . For an incomplete polynomial, the maximum degree of polynomial and the number of coefficients depend not only on the parameter  $M$  but also on the parameters  $i$  and  $j$  in (3). Detailed expressions of the components of matrix  $[B]$  are given in [3] for Mode I and mixed modes of II and III. A more convenient form for the mixed modes of II and III also can be found in [7].

Once the coefficients  $C$  are determined by solving (33) for given loadings on the crack surface, the stress-intensity factors corresponding to these loads are computed from the following equation [2].

For the Mode I problem,

$$\begin{aligned}K_I &= 8\mu \left( \frac{\pi}{a_1 a_2} \right)^{1/2} A^{1/4} \sum_{i=0}^1 \sum_{j=0}^1 \sum_{k=0}^M \sum_{\ell=0}^k (-2)^{2k+i+j} (2k+i+j+1)! \\ &\quad \times \frac{1}{a_1 a_2} \left( \frac{\cos \theta}{a_1} \right)^{2k-2\ell+i} \left( \frac{\sin \theta}{a_2} \right)^{2\ell+j} C_{3,k-\ell,\ell}^{(i,j)}\end{aligned}\quad (34)$$

where  $\theta$  is the elliptic angle and

$$A = a_1^2 \sin^2 \theta + a_2^2 \cos^2 \theta \quad (35)$$

For the mixed-mode problem of Modes II and III

$$K_{II} = 8\mu \left( \frac{\pi}{a_1 a_2} \right)^{1/2} A^{-1/4} \frac{1}{a_1 a_2} [H_1 a_2 \cos \theta + H_2 a_1 \sin \theta] \quad (36)$$

$$K_{III} = 8\mu \left( \frac{\pi}{a_1 a_2} \right)^{1/2} A^{-1/4} \frac{(1-\nu)}{a_1 a_2} [H_2 a_2 \cos \theta - H_1 a_1 \sin \theta] \quad (37)$$



in which

$$H_1 = \sum_{i=0}^1 \sum_{j=0}^1 \sum_{k=0}^M \sum_{\ell=0}^k (-2)^{2k+i+j} (2k+i+j+1)! \times \left( \frac{\cos \theta}{a_1} \right)^{2k-2\ell+i} \left( \frac{\sin \theta}{a_2} \right)^{2\ell+j} C_{1,k-\ell,\ell}^{(i,j)} \quad (38)$$

$$H_2 = \sum_{i=0}^1 \sum_{j=0}^1 \sum_{k=0}^M \sum_{\ell=0}^k (-2)^{2k+2-i-j} (2k+3-i-j)! \times \left( \frac{\cos \theta}{a_1} \right)^{2k-2\ell+1-i} \left( \frac{\sin \theta}{a_2} \right)^{2\ell+1-j} C_{2,k-\ell,\ell}^{(1-i,1-j)} \quad (39)$$

The VNA solution has been implemented by [8, 3] in a new finite-element alternating method for the solution of problems of embedded or surface flaws in complex structural geometries.

## 1.2 AN ELLIPTICAL CRACK IN A TRANSVERSELY ISOTROPIC SOLID

Recently, the authors in [9] have extended the VNA solution procedure to a transversely isotropic case, with arbitrary tractions on the face of an elliptical crack in a transversely isotropic solid, with the crack plane being at an arbitrary angle to the axis of elastic symmetry.

A material is said to be transversely isotropic when it possesses an axis of elastic symmetry such that the material is isotropic in the planes normal to this axis. Let  $\bar{z}$  be the direction of elastic symmetry. Then the stress-strain relations in the  $(\bar{x}, \bar{y}, \bar{z})$  system could be written as,

$$\begin{aligned} \sigma_{\bar{x}} &= C_{11} \frac{\partial u_{\bar{x}}}{\partial \bar{x}} + C_{12} \frac{\partial u_{\bar{x}}}{\partial \bar{y}} + C_{13} \frac{\partial u_{\bar{z}}}{\partial \bar{z}} \\ \sigma_{\bar{y}} &= C_{12} \frac{\partial u_{\bar{x}}}{\partial \bar{x}} + C_{11} \frac{\partial u_{\bar{y}}}{\partial \bar{y}} + C_{13} \frac{\partial u_{\bar{z}}}{\partial \bar{z}} \\ \sigma_{\bar{z}} &= C_{13} \left( \frac{\partial u_{\bar{x}}}{\partial \bar{x}} + \frac{\partial u_{\bar{y}}}{\partial \bar{y}} \right) + C_{33} \frac{\partial u_{\bar{z}}}{\partial \bar{z}} \\ \tau_{\bar{y}\bar{z}} &= C_{44} \left( \frac{\partial u_{\bar{z}}}{\partial \bar{y}} + \frac{\partial u_{\bar{y}}}{\partial \bar{z}} \right) \\ \tau_{\bar{z}\bar{x}} &= C_{44} \left( \frac{\partial u_{\bar{x}}}{\partial \bar{z}} + \frac{\partial u_{\bar{z}}}{\partial \bar{x}} \right) \\ \tau_{\bar{x}\bar{y}} &= \frac{1}{2} (C_{11} - C_{12}) \left( \frac{\partial u_{\bar{x}}}{\partial \bar{y}} + \frac{\partial u_{\bar{y}}}{\partial \bar{x}} \right) \end{aligned} \quad (40)$$

$(\sigma_{\bar{x}}, \sigma_{\bar{y}}, \sigma_{\bar{z}}, \tau_{\bar{y}\bar{z}}, \tau_{\bar{z}\bar{x}}, \tau_{\bar{x}\bar{y}})$  and  $(u_{\bar{x}}, u_{\bar{y}}, u_{\bar{z}})$  are stresses and displacement components in the  $(\bar{x}, \bar{y}, \bar{z})$  system and  $C_{ij}$  are elastic constants, as discussed in [10].

The displacement field  $u_{\bar{x}}$ ,  $u_{\bar{y}}$ , and  $u_{\bar{z}}$  is represented in terms of potential functions  $\phi_j$  ( $j = 1, 2, 3$ ), such that it satisfies the equilibrium equations expressed in terms of displacements, identically, as follows:

$$\begin{aligned} u_{\bar{x}} &= \frac{\partial}{\partial \bar{x}} (\phi_1 + \phi_2) - \frac{\partial \phi_3}{\partial \bar{y}} \\ u_{\bar{y}} &= \frac{\partial}{\partial \bar{y}} (\phi_1 + \phi_2) + \frac{\partial \phi_3}{\partial \bar{x}} \\ u_{\bar{z}} &= \frac{\partial}{\partial \bar{z}} (m_1 \phi_1 + m_2 \phi_2) \end{aligned} \quad (41)$$

$$m_j = \frac{C_{11}n_j - C_{44}}{C_{13} + C_{44}} = \frac{(C_{13} + C_{44})n_j}{C_{33} - C_{44}n_j}, \quad j = 1, 2 \quad (42)$$

The quantities  $n_1$  and  $n_2$  are the roots of the quadratic equation in  $n$

$$C_{11}C_{44}n^2 + [C_{13}(C_{13} + 2C_{44}) - C_{11}C_{33}]n + C_{33}C_{44} = 0 \quad (43)$$

and  $n_3$  is defined as:

$$n_3 = \frac{2C_{44}}{(C_{11} - C_{12})} \quad (44)$$

Introduction of the following modified coordinate systems  $(x_j, y_j, z_j)$  ( $j = 1, 2, 3$ )

$$\begin{aligned} x_j &= \bar{x} \\ y_j &= \bar{y} \cos \theta_j + \frac{\bar{z}}{\sqrt{n_j}} \sin \theta_j \\ z_j &= -\bar{y} \sin \theta_j + \frac{\bar{z}}{\sqrt{n_j}} \cos \theta_j \end{aligned} \quad (45)$$

yields the expressions of the elliptical crack in each modified coordinate system

$$\frac{x_j^2}{a_j^2} + \frac{y_j^2}{b_j^2} = 0 \quad \text{in } z_j = 0 \quad (j = 1, 2, 3 \text{ no sum}) \quad (46)$$

where

$$\begin{aligned} a_j &= a \\ b_j &= b \left( \cos \bar{\theta} \cos \theta_j + \frac{1}{\sqrt{n_j}} \sin \bar{\theta} \sin \theta_j \right) \end{aligned} \quad (47)$$

and

$$\tan \theta_j = \frac{1}{\sqrt{n_j}} \tan \bar{\theta} \quad (48)$$

Here,  $\bar{\theta}$  is the angle between the physical crack axis ( $z$ ) and the material axis ( $\bar{z}$ ).

Now, each of the potentials  $\phi_j$  ( $j = 1, 2, 3$ ) can be expressed by the harmonic equations in a set of coordinate  $(x_j, y_j, z_j)$  ( $j = 1, 2, 3$ ), as:

$$\left( \frac{\partial^2}{\partial x_j^2} + \frac{\partial^2}{\partial y_j^2} + \frac{\partial^2}{\partial z_j^2} \right) \phi_j = 0 \quad (j = 1, 2, 3 \text{ no sum on } j) \quad (49)$$

Thus, to solve the problem on hand, appropriate potential functions  $\phi_j$  ( $j = 1, 2, 3$ ) each in a different set of coordinates  $(x_j, y_j, z_j)$  ( $j = 1, 2, 3$ ) will now be assumed. The necessary ellipsoidal coordinates  $\xi_1^j, \xi_2^j, \xi_3^j$  ( $j = 1, 2, 3$ ) for a point in the  $x_j, y_j, z_j$  ( $j = 1, 2, 3$ ) coordinate system are given by the roots of the cubic equation

$$\omega_j(\xi^j) = 0 \quad (j = 1, 2, 3 \text{ no sum on } j) \quad (50)$$

where

$$\omega_j(\xi^j) = 1 - \frac{x_j^2}{a^2 + \xi^j} - \frac{y_j^2}{b_j^2 + \xi^j} - \frac{z_j^2}{\xi^j} \quad (51)$$

and

$$-a^2 \leq \xi_1^j \leq -b_j^2 \leq \xi_2^j \leq 0 \leq \xi_3^j < \infty \quad (52)$$

Appropriate expressions of the potentials  $\phi_j$  for the present case of transverse isotropy as discussed above, are assumed as:

$$\phi_j = \sum_{\ell=0}^M \sum_{k=0}^{\ell} B_{\ell-k,k}^j F_{\ell-k,k}^j \quad (53)$$

where,  $B_{\ell-k,k}^j$  are unknown coefficients to be determined,  $M$  is the highest order polynomial terms considered and  $F_{\ell-k,k}^j$  is defined as:

$$F_{k\ell}^j = \frac{\partial^{k+\ell}}{\partial x_j^k \partial y_j^\ell} \int_{\xi_3^j}^{\infty} [\omega_j(s)]^{k+\ell+1} \frac{ds}{\sqrt{Q_j(s)}} \quad (54)$$

As can be easily noticed, the above equation is basically the same as Eq. (5) or Eq. (10) in the VNA solution procedure for real  $n_1$  and  $n_2$  Eq. (44). Therefore, the VNA solution procedure can be used to obtain the complete general solution for an transversely isotropic material. However, for complex  $n_1$  and  $n_2$  analytic continuation from the real axis to the complex plane has to be carried out. This solution is given in [9] in more detail.

### 1.3 A CIRCULAR CRACK IN AN ISOTROPIC SOLID

The analytical general solution for a circular crack in an infinite isotropic elastic solid, subject to arbitrary crack-face tractions, is briefly summarized here. The solution was revisited in [11], based on the Fourier-Hankel transform technique developed in [12], which was later generalized in [13]. Although the authors in [13] have derived the general solution for this type of problem, certain portions of the mixed-mode solution were lacking in their final results. Thus, the complete form of the general solution has been recently rederived [11] as follows:

For a penny-shaped crack embedded in an infinite 3-D elastic body, we need to solve the following mixed boundary value problems.

#### Mode I

$$\begin{aligned} \sigma_{rz}(r, \theta, 0) = \sigma_{\theta z}(r, \theta, 0) &= 0 & r &\geq 0; & 0 \leq \theta \leq 2\pi \\ \sigma_{zz}(r, \theta, 0) &= p_1(r, \theta) & 0 \leq r < a; & & 0 \leq \theta \leq 2\pi \\ u_z(r, \theta, 0) &= 0 & r > a; & & 0 \leq \theta \leq 2\pi \end{aligned} \quad (55)$$

( $a$  is the crack radius)

#### Modes II and III

$$\begin{aligned} \sigma_{zz}(r, \theta, 0) &= 0 & r &\geq 0 & 0 \leq \theta \leq 2\pi \\ \sigma_{rz}(r, \theta, 0) &= p_2(r, \theta) & 0 \leq r < a & & 0 \leq \theta \leq 2\pi \\ \sigma_{\theta z}(r, \theta, 0) &= p_3(r, \theta) & 0 \leq r < a & & 0 \leq \theta \leq 2\pi \\ u_r(r, \theta, 0) = u_\theta(r, \theta, 0) &= 0 & r > a & & 0 \leq \theta \leq 2\pi \end{aligned} \quad (56)$$

where  $p_\alpha(r, \theta)$  ( $\alpha = 1, 2, 3$ ) are given functions describing the distribution of the loads applied to the crack surface.

An appropriate solution for this boundary-value problem can be obtained by expressing the displacement components in terms of three harmonic functions  $\phi_\alpha$  ( $\alpha = 1, 2, 3$ ), as

$$\begin{aligned} u_r &= (1-2\nu) \frac{\partial \phi_1}{\partial r} + z \frac{\partial^2 \phi_1}{\partial r \partial z} + \frac{2}{r} \frac{\partial \phi_2}{\partial \theta} + 2(1-\nu) \frac{\partial \phi_3}{\partial r} + z \frac{\partial^2 \phi_3}{\partial r \partial \theta} \\ u_\theta &= (1-2\nu) \frac{1}{r} \frac{\partial \phi_1}{\partial \theta} + \frac{z}{r} \frac{\partial^2 \phi_1}{\partial \theta \partial z} - 2 \frac{\partial \phi_2}{\partial r} + 2(1-\nu) \frac{1}{r} \frac{\partial \phi_3}{\partial \theta} + \frac{z}{r} \frac{\partial^2 \phi_3}{\partial \theta \partial z} \\ u_z &= -2(1-\nu) \frac{\partial \phi_1}{\partial z} + z \frac{\partial^2 \phi_1}{\partial z^2} - (1-2\nu) \frac{\partial \phi_3}{\partial z} + z \frac{\partial^2 \phi_3}{\partial z^2} \end{aligned} \quad (57)$$



The corresponding stress components in terms of the potential functions can be easily obtained by the above equations through the use of the strain-displacement and the stress-strain relations. From the above equations, it is seen that  $\phi_1$  is related with mode I, and  $\phi_2$  and  $\phi_3$  are related with the mixed mode of II and III.

In order to express general loadings, the applied loads  $p_\alpha(r, \theta)$  are expressed by Fourier series as follows:

$$p_\alpha(r, \theta) = \sum_{n=0}^{\infty} \frac{\cos n\theta \cdot A_{\alpha n}(r)}{\sin n\theta \cdot B_{\alpha n}(r)} \quad (\alpha = 1, 2, 3) \quad (58)$$

In order to solve the proposed problem, the potential functions are represented by the Fourier-Hankel transform:

$$\phi_\alpha(r, \theta, z) = \sum_{n=0}^{\infty} \frac{\cos n\theta}{\sin n\theta} \int_0^\infty \frac{C_{\alpha n}(s)}{D_{\alpha n}(s)} \frac{1}{s} J_n(rs) e^{-sz} ds \quad (\alpha = 1, 2, 3) \quad (59)$$

The substitution of Eqs. (58) and (59) into Eqs. (55), (56) and (57) yields the following relations:

Mode I

$$\begin{aligned} C_{1n}(s) &= -\frac{1}{\mu} \sqrt{\frac{s}{2\pi}} \int_0^a J_{n+1/2}(st) \frac{dt}{t^{n-1/2}} \int_0^t \frac{r^{n+1} A_{1n}(r) dr}{(t^2 - r^2)^{1/2}} \\ D_{1n}(s) &= -\frac{1}{\mu} \sqrt{\frac{s}{2\pi}} \int_0^a J_{n+1/2}(st) \frac{dt}{t^{n-1/2}} \int_0^t \frac{B_{1n}(r) dr}{(t^2 - r^2)^{1/2}} \end{aligned} \quad (60)$$

Mode II and III

$$\begin{aligned} C_{20}(s) &= -\frac{1}{\mu} \sqrt{\frac{s}{2\pi}} \int_0^a J_{3/2}(st) \frac{dt}{t^{1/2}} \int_0^t \frac{r^2 A_{30}(r)}{(t^2 - r^2)^{1/2}} dr \\ C_{30}(s) &= \frac{1}{\mu} \sqrt{\frac{s}{2\pi}} \int_0^a J_{3/2}(st) \frac{dt}{t^{1/2}} \int_0^t \frac{r^2 A_{20}(r)}{(t^2 - r^2)^{1/2}} dr \end{aligned} \quad (61)$$

$$\begin{aligned} C_{2n}(s) &= \sqrt{s} \int_0^a [(\nu - 1) \Phi_1^*(t) J_{n-1/2}(st) + \Phi_2^*(t) J_{n+3/2}(st)] dt, \quad n \geq 1 \\ C_{3n}(s) &= \sqrt{s} \int_0^a [\Phi_1(t) J_{n-1/2}(st) + \Phi_2(t) J_{n+3/2}(st)] dt, \quad n \geq 1 \\ D_{2n}(s) &= \sqrt{s} \int_0^a [(1 - \nu) \Phi_1(t) J_{n-1/2}(st) - \Phi_2(t) J_{n+3/2}(st)] dt, \quad n \geq 1 \\ D_{3n}(s) &= \sqrt{s} \int_0^a [\Phi_1^*(t) J_{n-1/2}(st) + \Phi_2^*(t) J_{n+3/2}(st)] dt, \quad n \geq 1 \end{aligned} \quad (62)$$

where,

$$\begin{aligned} \Phi_1(t) &= \frac{-t^{-n+3/2}}{(2 - \nu)\mu\sqrt{2\pi}} \int_0^t \frac{r^n [A_{2n}(r) - B_{3n}(r)]}{(t^2 - r^2)^{1/2}} dr \\ \Phi_1^*(t) &= \frac{-t^{-n+3/2}}{(2 - \nu)\mu\sqrt{2\pi}} \int_0^t \frac{r^n [A_{3n}(r) + B_{2n}(r)]}{(t^2 - r^2)^{1/2}} dr \\ \Phi_2(t) &= \frac{\nu}{2} \Phi_1(t) + \frac{t^{-n-1/2}}{2\mu\sqrt{2\pi}} \left\{ \frac{(1 + 2n)\nu}{2 - \nu} \int_0^t r^n [A_{2n}(r) - B_{3n}(r)] \right. \\ &\quad \left. \times (t^2 - r^2)^{1/2} dr + \int_0^t \frac{r^{n+2} [A_{2n}(r) + B_{3n}(r)]}{(t^2 - r^2)^{1/2}} dr \right\} \end{aligned}$$

$$\begin{aligned}\Phi_2^*(t) = & \frac{\nu}{2}\Phi_1^*(t) + \frac{t^{-n-1/2}}{2\mu\sqrt{2\pi}} \left\{ \frac{(1+2n)\nu}{2-\nu} \int_0^t r^n [A_{3n}(r) + B_{2n}(r)] \right. \\ & \times (t^2 - r^2)^{1/2} dr + \left. \int_0^t \frac{r^{n+2}[B_{2n}(r) - A_{3n}(r)]}{(t^2 - r^2)^{1/2}} dr \right\}\end{aligned}\quad (63)$$

Without going into details, the stress intensity factors for all the modes are given by

$$\begin{aligned}K_I &= -\frac{2}{\sqrt{\pi}} \sum_{n=0}^{\infty} \frac{\cos n\theta}{\sin n\theta} \frac{1}{a^{n+1/2}} \int_0^a \frac{A_{1n}(r)}{B_{1n}(r)} \frac{r^{n+1}}{(a^2 - r^2)^{1/2}} dr \\ K_{II} &= -\frac{2}{\sqrt{\pi}a^{3/2}} \int_0^a \frac{r^2 A_{20}(r)}{(a^2 - r^2)^{1/2}} dr + \frac{2\mu\sqrt{2}}{a} \sum_{n=1}^{\infty} [\Phi_1(a) - \Phi_2(a)] \cos n\theta \\ &\quad - \frac{2\mu\sqrt{2}}{a} \sum_{n=1}^{\infty} [(1-\nu)\Phi_1^*(a) + \Phi_2^*(a)] \sin n\theta \\ K_{III} &= -\frac{2\mu\sqrt{2}}{a} \sum_{n=1}^{\infty} [(1-\nu)\Phi_1(a) + \Phi_2(a)] \sin n\theta \\ &\quad - \frac{2}{\sqrt{\pi}a^{3/2}} \int_0^a \frac{r^2 A_{30}(r)}{(a^2 - r^2)^{1/2}} dr + \frac{2\mu\sqrt{2}}{a} \sum_{n=1}^{\infty} [\Phi_1^*(a) - \Phi_2^*(a)] \cos n\theta\end{aligned}\quad (64)$$

## 2 FINITE-ELEMENT AND BOUNDARY-ELEMENT ALTERNATING METHODS FOR 2D AND 3D CRACK PROBLEMS

Consider a 2 or 3 dimensional homogeneous solid containing multiple cracks, as shown in Fig. 1, and assume that the crack-faces are traction-free. In the alternating technique [14], and [15], the stresses in the *uncracked* body are first analyzed, by using a numerical/analytical method such as the finite element/boundary element method, for the given system of external loading. In order to assess the effect of the crack, the tractions at the locations of the cracks in the otherwise uncracked body must be erased.

Assume for simplicity, for now, that there is a single crack. Thus, one has to deal with the problem of a *finite body* with a crack, the faces of which are subject to arbitrary tractions, and the outer boundaries are traction free. Denoting by  $S_c$  the crack-face, and by  $S_\sigma$  the external boundary of the *finite-body*  $V$ , this problem may be posed as:

$$\text{Solve : } L(u) = 0 \text{ in } V ; \quad B(u) = \bar{t} \text{ at } S_c ; \quad B(u) = 0 \text{ at } S_\sigma \quad (65)$$

where  $u$  are the displacements, and  $L$  and  $B$  are the appropriate differential operators. To solve problem (65) by using the alternating method, the finite body is replaced by an *infinite body*, with stresses going to zero at infinity. The problem of an infinite homogeneous body containing a crack, the faces of which are subject to tractions  $\bar{t}$  as in (65), does have an *analytical* solution often times, and hence need not be solved numerically. The far-field stresses from this *infinite body* solution do not satisfy the condition  $B(u) = 0$  at  $S_\sigma$ . Thus, the residual tractions at the boundaries of the finite-body are erased, by first solving an *uncracked* body with these residual boundary tractions, and then erasing the tractions at the location of the crack in the uncracked body. This last problem of erasing crack-face tractions is similar to problem (65) above. This iterative loop is continued until the analytical solution for the infinite body satisfies also the zero-traction condition,  $B(u) = 0$  at  $S_\sigma$ , of the finite body. Thus, in the alternating method, problem (65) is recast as:

$$L(u) = 0 \text{ in } V_\infty ; \quad B(u) = \bar{t}^{(i)} \text{ at } S_c ; \quad \text{and } B(u) = \bar{t}_\sigma^{(i)} \text{ at } S_\sigma \quad (66)$$

where  $i$  denotes the  $i$ th iteration, and  $V_\infty$  is the infinite domain. The iteration continues until  $\bar{t}_\sigma^{(i)} = 0$ . Let  $\bar{t}^{(i)}$  be the converged value at  $S_c$ , corresponding to which the stress-intensity factors are determined from

the analytical infinite-body solution [15,1]. For multiple cracks, the alternating method is analogous; and is represented schematically in Figs. 1 and 2. In a variety of practical cases, this alternating method has been established to be a very simple and cost-effective tool of analysis [16].

- Alternating Technique for Multiple 2D or 3D Embedded/Surface Cracks

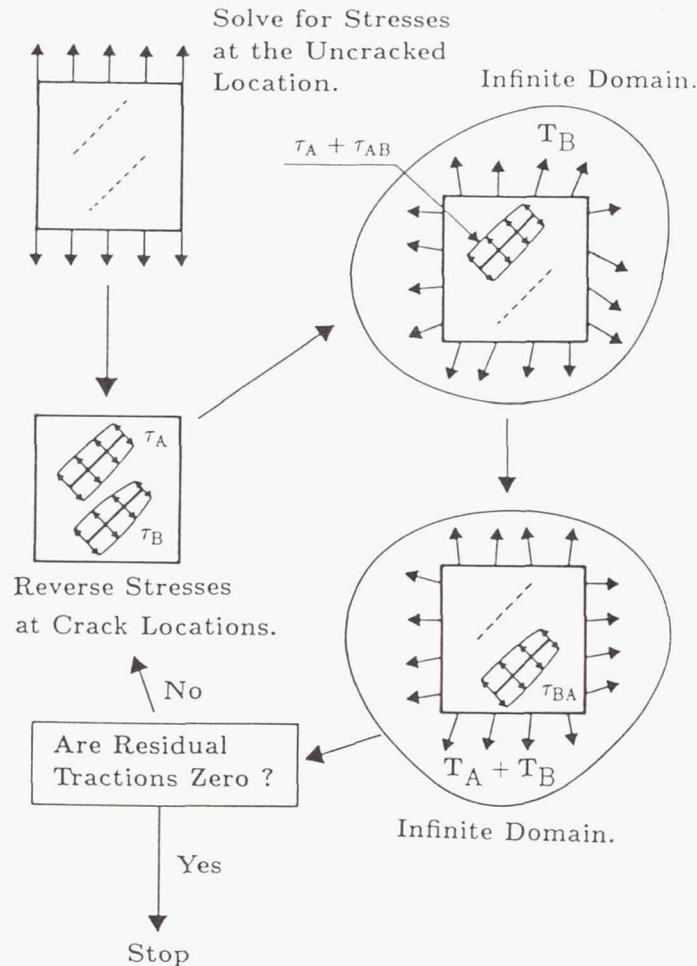


Fig. 1

Henceforth we assume, for convenience, that a finite-element method is used. The steps involved in the finite-element alternating method for an embedded crack in a finite body are described below:

1. Solve the *uncracked finite body* under the prescribed external loads by using the finite element method. The uncracked body has the same geometry as the given problem except for the crack.
2. Using the finite element solution, compute the stresses at the location of the crack.
3. Compare the residual stresses calculated in step 2 with a permissible stress magnitude.
4. To create traction-free crack faces as in the given problem, reverse the residual stress at the location of the crack as computed in Step 2 and "least-squares fit" them to polynomials.
5. Obtain the analytical solution to the infinite body with the crack subject to the polynomial loading as in Step 4.



6. Calculate the stress intensity factors for the current iteration, using the above analytical solution.
7. Calculate the residual stresses on external surfaces at the body due to the applied loads on crack-faces, as in step 4. To satisfy the given traction boundary conditions, at the external boundaries, reverse the residual stresses on the external surfaces of the body, and calculate the equivalent nodal forces.
8. Consider the nodal forces in step 7 as externally applied loads acting on the uncracked body.

Repeat all steps in the iteration process until the residual stress on the crack surface becomes negligible. To obtain the final solution, add the stress intensity factors for all iterations.

Since the alternating method is iterative in nature, the finite-element equations may, in general, have to be solved repeatedly for different applied loads, while keeping the stiffness matrix the same. To save computational time, special computational techniques were implemented in [3]. These are explained below.

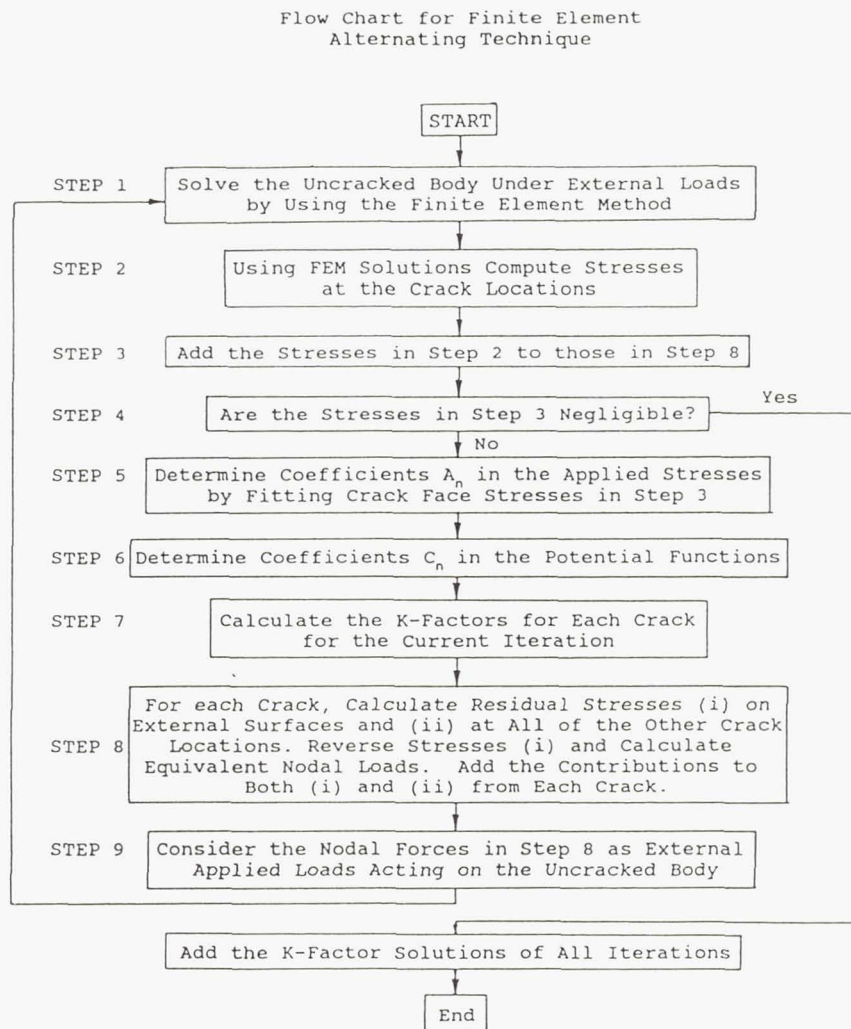


Fig. 2

As seen from above, for the finite-element alternating method, we need to solve the following type of finite-element equations:

$$[K][q^0, q^1, \dots, q^n] = [Q^0, Q^1, \dots, Q^n] \quad (67)$$

and

$$Q^i = Q^i(q^{i-1}); \quad i = 1, 2, \dots, n \quad (68)$$

in which the superscript denotes the cycle of iteration,  $[K]$  is the global (assembled) stiffness matrix of the *uncracked* body, and remains the same during the iteration process, and  $q^i$  is the nodal displacement vector for  $i$ th iteration.  $Q^i$  is the nodal force vector for the  $i$ th iteration and depends on the solution for the previous iteration  $q^{i-1}$  as expressed by Eq. (68).

An efficient equation solver OPTBLOK developed in [17] may be used to save computational time in solving Eq. (67). The solution algorithm is divided into three parts, i.e. (i) reduction of stiffness matrix, (ii) reduction of load vector, and (iii) back substitution. In OPTBLOK the reduction of stiffness matrix is done only once, although the reduction of load vector and back substitution may be repeated for any number of load cases. Thus, denoting the CPU time for each part by  $T_1$ ,  $T_2$ , and  $T_3$ , respectively, the total CPU time  $T$  in solving Eq. (67) using OPTBLOK can be expressed as

$$T = T_1 + (n + 1)(T_2 + T_3) = (T_1 + T_2 + T_3) + n(T_2 + T_3) \quad (69)$$

where  $n$  is the total number of iterations. Since  $T_1$  is much larger than  $(T_2 + T_3)$ , a substantial reduction in computational time, compared with the case in which Eq. (67) is solved for each iteration [i.e.  $T^* = (n+1)(T_1+T_2+T_3)$ ], may be expected. To illustrate this situation, we consider the example of a set of linear equations with the number of equations of 1960, and half bandwidth of 200, wherein the CPU time for reduction of load vector and back substitution was about 5.6% of the total CPU time ( $T_2 + T_3 \simeq 0.056T$ ). Since, for a typical problem, the present alternating method needs three iterations ( $n = 3$ ), the additional cost in this case is only about 16.8%, which is considerably smaller than the 300% in the case when Eq. (67) is solved for each iteration.

An efficient procedure was also devised for the calculation of the nodal forces required in step 7 (see also Eq. (68)). In general, the stress field in a general solution can be expressed by

$$\sigma = PC \quad (70)$$

where  $P$  is the basis function matrix for stresses, and  $C$  is the vector of unknown coefficients in the general solution which will be determined in step 5. Then, the equivalent nodal forces in step 7 can be computed through:

$$Q_m = -G_m C \quad (71)$$

and

$$G_m = \int_{S_m} N^t n P dS \quad (72)$$

where  $m$  denotes the number for a finite element,  $Q_m$  are nodal forces,  $N$  is the matrix of the element shape functions,  $n$  is the matrix of the normal direction cosines. Although the matrix  $P$  has the singularity of order  $1/\sqrt{r}$  at the crack-front, the functions in  $P$  decay very rapidly with the distance from the crack-front. Thus, the matrices  $G_m$  are calculated only at the external boundary-surface elements which satisfy the condition  $r_{\min} < 5a_1$ , where  $r_{\min}$  is the distance of the closest nodal point of each external boundary-surface element from the center of the ellipse and  $a_1$  is the semi-major axis of the ellipse.

To save computation time, the  $G_m$  matrix can be calculated only once prior to the start of the iteration process. Thus, the equivalent nodal forces  $Q^i$  in each iteration can be evaluated without integration.

## 2.2 3D ALTERNATING TECHNIQUES FOR PART-ELLIPTICAL SURFACE CRACKS

The VNA solution given in 2.1 serves as Solution 1 required in the alternating technique.

Now, some comments concerning the solution of *surface flaw* problems in finite bodies, through the present procedure, are in order. Since the analytical solution for an elliptical crack in an infinite solid is implemented as solution (1), it is necessary to define the residual stresses over the entire crack plane including the fictitious portion of the crack which lies outside of the finite body. Moreover, it is well known that the



accuracy of the "least-squares" function interpolation inside the interpolated region can be increased with the number of polynomial terms; however, the interpolating curve may change drastically outside the region of interpolation. For these reasons, in [3] numerical experimentation was carried out to arrive at an optimum pressure distribution on the crack surface extended into the fictitious region. For a semi-elliptical crack which lies in the region of  $-a_1 \leq x_1 \leq a_1$  and  $0 \leq x_2 \leq a_2$ , it was concluded that the fictitious pressure, which, for the region of  $-a_2 \leq x_2 \leq 0$ , remains constant in the  $x_2$  direction but varies in the  $x_1$  direction, gives the best result among the several numerical experiments performed in [3], even though the results for other types of assumed pressure in the fictitious region differed only slightly ( $\pm 2\%$ ).

This procedure of fictitious pressure distribution for a semi-elliptical surface crack was successfully used on the analyses of surface cracks, in finite-thickness plates subject to remote tension as well as remote bending [3], and in pressure vessels [18].

Based on the studies in [19], the following "fictitious" stress distribution is recommended for quarter-elliptical surface cracks. For the first quadrant ( $x_1, x_2 \geq 0$ ) (namely, the actual surface crack), the residual stress can be calculated by the finite element method and is a function of the coordinates  $x_1$  and  $x_2$ . For the other quadrants, the fictitious residual stress is defined as

$$\sigma_{33}^R = \begin{cases} \sigma_{33}^R(0, x_2) & \text{for the second quadrant } (x_1 \leq 0, x_2 \geq 0) \\ \sigma_{33}^R(0, 0) & \text{for the third quadrant } (x_1 \leq 0, x_2 \leq 0) \\ \sigma_{33}^R(x_1, 0) & \text{for the fourth quadrant } (x_1 \geq 0, x_2 \leq 0) \end{cases} \quad (73)$$

The above alternating method has been successfully applied to the problem of semi-elliptical surface flaws in plates subjected to tension and bending [8, 3], semi-elliptical surface flaws in the meridional direction at the outer and inner surfaces of pressurized thick and thin cylindrical vessels [18], quarter-elliptical surface flaws emanating from pin-holes in attachment lugs [19], *multiple* coplanar embedded elliptical flaws in an infinite solid subject to arbitrary crack-face tractions [20], and *multiple* semi-elliptical surface flaws in the meridional as well as circumferential directions in cylindrical pressure vessels [14, 21].

The nature of singularity at the point where the crack-front intersects the free-surface is still not yet completely understood. The consensus emerging from the literature of a weaker singularity (than  $1/\sqrt{r}$ ) at a normal crack/surface interaction has been corroborated recently, in [22]. These authors present two independent numerical analysis techniques for the investigation of some global crack/surface interaction problems. They summarize their findings, thus: "the decays in the energy release rates found as the free surface is approached in the various problems treated are probably not significant from a fracture toughness testing point of view and not of major consequence in cyclic life calculations, although there are some indications that this may not be the case if near-surface residual stress fields are present; and that these variations in energy release rate can be compensated for by relatively minor perturbations in crack-front profiles".

Thus the results obtained by the above finite element alternating method based on the VNA solution may be thought of as being of adequate accuracy for most engineering applications. Recently, [23] a more efficient alternating method for the analysis of a group of interacting multiple elliptical cracks was developed, by taking account of geometrical symmetries of crack shapes and location in conjunction with the symmetry of the VNA solution.

Intensive studies of the performance of the finite-element alternating method have been made in [24] for small surface and corner cracks, and in [25] for a part-elliptical surface crack in a cylinder. From the performance studies of the finite-element alternating method, [25] summarized the attractive features of the alternating technique as follows:

- (i) The method models only the uncracked solid with finite elements: hence, no special modeling of the crack front is required. In addition, the finite element mesh at the location of the crack, in the uncracked solid, can be completely arbitrary in geometry.
- (ii) The method uses the closed-form solution for a crack in an infinite solid which can accommodate arbitrary tractions on the crack surfaces and, therefore, can handle complex loading conditions.



(iii) The stress-intensity factors, including the individual modes, are obtained as part of the solution, in an analytical form, and, hence, post-processing of the output data, as is usually done in the finite-element method, is not needed.

(iv) Several crack configurations could be analysed with a single arbitrary mesh idealization of the uncracked solid, whereas the conventional finite-element method requires a different mesh idealization of the cracked structure for each crack configuration. Thus, this method can efficiently generate very accurate stress-intensity factor weight functions or influence functions, for a variety of crack aspect ratios, in a single computer run.

The above applications of the alternating technique were limited to mode I cases. Recently, a *mixed-mode* alternating finite-element technique in conjunction with the VNA solution (with further improvements in algebraic details), has been developed in [7]. They evaluated the polynomial influence functions for an infinite solid with an elliptical crack subject to shear loading, and for a cantilever beam with a semi-elliptical surface crack subject to end load.

Applications of the finite-element alternating method have been made in [26, 27, 28], for fracture mechanics analyses of various offshore structural components, such as stiffened plate and shells, tethers, or risers. A recent literature survey [29] pointed out that the alternating method is most efficient for stress intensity factor analyses of *planar* surface or embedded flaws in complex geometries such as intersecting tubular structures, etc.

### 2.3 2 D ALTERNATING TECHNIQUES FOR LINE CRACKS

As mentioned earlier, the general solution for a crack subject to arbitrary crack-face tractions i.e. Solution 1 is required. The general solution for an infinite 2D anisotropic body, developed along the lines of [30], is given below. Following the solution procedure in [31], the stress and displacement field can be expressed in terms of two potential  $\phi$  and  $\psi$  as follows:

$$\begin{aligned}\tau_{xx} &= 2 \operatorname{Re}[s_1^2 \phi'(z_1) + s_2^2 \psi'(z_2)] \\ \tau_{yy} &= 2 \operatorname{Re}[\phi'(z_1) + \psi'(z_2)] \\ \tau_{xy} &= -2 \operatorname{Re}[s_1 \phi'(z_1) + s_2 \psi'(z_2)] \\ u &= 2 \operatorname{Re}[p_1 \phi(z_1) + p_2 \psi(z_2)] \\ v &= 2 \operatorname{Re}[q_1 \phi(z_1) + q_2 \psi(z_2)]\end{aligned}\quad (74)$$

where

$$\begin{aligned}s_1 &= \mu_1 = \alpha_1 + i\beta_1, & s_2 &= \mu_2 = \alpha_2 + i\beta_2 \\ \mu_3 &= \bar{\mu}_1 & \mu_4 &= \bar{\mu}_2 \\ z_1 &= x + s_1 y & z_2 &= x + s_2 y & s_1 &\neq s_2\end{aligned}\quad (75)$$

where  $\alpha_j, \beta_j$  ( $j = 1, 2$ ) are real constants.  $\mu_j$  ( $j = 1, 2, \dots, 4$ ) are the roots of the characteristic equation

$$a_{11}\mu_j^4 - 2a_{16}\mu_j^3 + (2a_{12} + a_{66})\mu_j^2 - 2a_{26}\mu_j + a_{22} = 0 \quad (76)$$

where  $a_{ij}$  ( $i, j = 1, 2, \dots, 6$ ) are the material constants of generalized Hooke's law

$$\begin{aligned}\varepsilon_x &= a_{11}\tau_{xx} + a_{12}\tau_{yy} + a_{16}\tau_{xy} \\ \varepsilon_y &= a_{12}\tau_{xx} + a_{22}\tau_{yy} + a_{26}\tau_{xy} \\ \gamma_{xy} &= a_{16}\tau_{xx} + a_{26}\tau_{yy} + a_{66}\tau_{xy}\end{aligned}\quad (77)$$

The other constants in Eq. (91) are defined as:

$$\begin{aligned}p_1 &= a_{11}s_1^2 + a_{12} - a_{16}s_1 & p_2 &= a_{11}s_2^2 + a_{12} - a_{16}s_2 \\ q_1 &= \frac{a_{12}s_1^2 + a_{22} - a_{26}s_1}{s_1} & q_2 &= \frac{a_{12}s_2^2 + a_{22} - a_{26}s_2}{s_2}\end{aligned}\quad (78)$$

Suppose a line crack on  $y = 0$   $|x| \leq a$  in an infinite plane is inflated by equal and opposite tractions, over the faces of the crack, given by

$$\tau_{yy} - i\tau_{xy} = -[p(t) + is(t)], \quad |t| \leq a \quad (79)$$

with zero tractions at infinity. Then the potential functions can be written as below:

$$\begin{aligned} \phi'(z_1) &= \phi'_1(z_1) + \phi'_2(z_1) \\ \psi'(z_2) &= \psi'_1(z_2) + \psi'_2(z_2) \end{aligned} \quad (80)$$

where

$$\begin{aligned} \left(\frac{s_2 - s_1}{s_2}\right) \phi'_1(z_1) &= -\frac{X(z_1)}{2\pi i} \int_{-a}^a \frac{p(t)dt}{[X(t)]^+(t - z_1)} \\ \left(\frac{s_1 - s_2}{s_1}\right) \psi'_1(z_2) &= -\frac{Y(z_2)}{2\pi i} \int_{-a}^a \frac{p(t)dt}{[Y(t)]^+(t - z_2)} \end{aligned} \quad (81)$$

where

$$\begin{aligned} X(z_1) &= (z_1 + a)^{-1/2}(z_1 - a)^{-1/2} \\ Y(z_2) &= (z_2 + a)^{-1/2}(z_2 - a)^{-1/2} \end{aligned} \quad (82)$$

and

$$\begin{aligned} (s_1 - s_2)\phi'_2(z_1) &= -\frac{X(z_1)}{2\pi i} \int_{-a}^a \frac{s(t)dt}{[X(t)]^+(t - z_1)} \\ (s_2 - s_1)\psi'_2(z_2) &= -\frac{Y(z_2)}{2\pi i} \int_{-a}^a \frac{s(t)dt}{[Y(t)]^+(t - z_2)} \end{aligned} \quad (83)$$

We approximate the applied crack-face tractions in the form [30]

$$p(t) + is(t) = -\sum_{n=1}^N b_n U_{n-1}(t) \quad |t| \leq a \quad (84)$$

where  $U_{n-1}(t)$  is the Chebyshev polynomials of the second kind and is defined as

$$U_n = \sin[(n+1)\theta]/\sin\theta \quad t = a \cos\theta \quad (85)$$

It could be easily shown that:

$$\begin{aligned} 2\phi'(z_1) &= \left(\frac{s_2}{s_2 - s_1}\right) \sum_{n=1}^N c_n G_{n-1}(z_1) + \left(\frac{1}{s_1 - s_2}\right) \sum_{n=1}^N d_n G_{n-1}(z_1) \\ 2\psi'(z_2) &= \left(\frac{s_1}{s_1 - s_2}\right) \sum_{n=1}^N c_n G_{n-1}(z_2) + \left(\frac{1}{s_2 - s_1}\right) \sum_{n=1}^N d_n G_{n-1}(z_2) \end{aligned} \quad (86)$$

where

$$\begin{aligned} c_n &= \text{real}(b_n) \\ d_n &= i(\text{imag}(b_n)) \end{aligned}$$

$$\begin{aligned}
2\phi(z_1) &= \left( \frac{s_2}{s_2 - s_1} \right) \sum_{n=1}^N c_n \frac{R_n(z_1)}{n} + \left( \frac{1}{s_1 - s_2} \right) \sum_{n=1}^N d_n \frac{R_n(z_1)}{n} \\
2\psi(z_2) &= \left( \frac{s_1}{s_1 - s_2} \right) \sum_{n=1}^N c_n \frac{R_n(z_2)}{n} + \left( \frac{1}{s_2 - s_1} \right) \sum_{n=1}^N d_n \frac{R_n(z_2)}{n}
\end{aligned} \tag{87}$$

and

$$G_{n-1}(z_\alpha) = -(z_\alpha^2 - a^2)^{-1/2} R_n(z_\alpha) \quad (\alpha = 1, 2) \tag{88}$$

$$R_n(z_\alpha) = a \{ z_\alpha/a - (z_\alpha^2/a^2 - 1)^{1/2} \}^n \quad (\alpha = 1, 2) \tag{89}$$

The stress intensity factors  $K_j$  ( $j = I, II$ ) are defined in a manner consistent with those for isotropic materials.

$$\begin{aligned}
K_I &= 2\sqrt{2\pi} \left( \frac{s_2 - s_1}{s_2} \right) \lim_{z_1 \rightarrow a} (z_1 - a)^{1/2} \phi_1'(z_1) \\
K_{II} &= 2\sqrt{2\pi} (s_2 - s_1) \lim_{z_1 \rightarrow a} (z_1 - a)^{1/2} \phi_2'(z_1)
\end{aligned} \tag{90}$$

It is easy to show that

$$\begin{aligned}
K_I - iK_{II} &= -\sqrt{\pi a} \sum_{n=1}^N b_n \quad \text{for } x = a \\
K_I - iK_{II} &= \sqrt{\pi a} \sum_{n=1}^N (-1)^n b_n \quad \text{for } x = -a
\end{aligned} \tag{91}$$

### 3 THE CURRENT STATE-OF-THE-ART: "THE DISPLACEMENT COMPATIBILITY APPROACH" FOR CRACKED, STIFFENED FUSELAGE PANELS

The methods that are currently used for the fracture analysis of stiffened panels are based on the so-called "displacement-compatibility approach" as developed in [32, 33], and others see, for instance, [34]. The "displacement-compatibility" approach is schematically illustrated in Fig. 3. This method can be classified as a general "flexibility matrix" approach in structural mechanics. In this approach, the effects of the fasteners on the sheet are represented by a series of concentrated forces on the sheet, which may contain multiple cracks, and the sheet is subjected, in addition, to the far-field hoop stress [labelled henceforth as problem A].

The stringers, in turn, are acted upon by a series of concentrated forces, equal in magnitude and opposite in direction to those in problem A. The displacement compatibility between the sheet and the stiffener, at each fastener location, is enforced, by taking into account the fastener flexibility, as:

$$v_{sheet} = v_{stiffener} + v_{fastener} \tag{92}$$

(at each fastener location)

Problem A above is again broken-up, by the superposition principle, into 3 problems, labelled here as problems B, C and D, respectively. Problem B is that of a sheet, subjected to hoop membrane stress, and containing multiple cracks. Problem C is that of the uncracked sheet subjected to a series of concentrated forces as in Problem A, at the fastener locations. Problem D is that of a sheet containing multiple cracks, each of which is subjected to tractions that are equal in magnitude, but opposite in direction, to the residual tractions existing at the locations of the cracks in an otherwise uncracked sheet, as in Problem C.

$$v_{sheet} = v_B + v_C + v_D \tag{93}$$

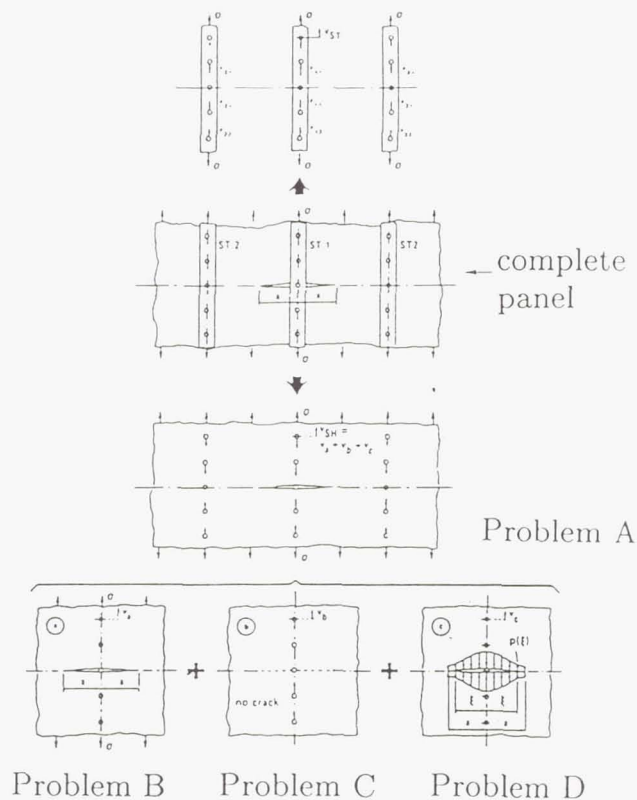


(at each fastener location)

or

$$v_b + v_C + v_D = V_{stiffener} + v_{fastener} \quad (94)$$

(at each fastener location)



## DISPLACEMENT COMPATIBILITY METHOD

Fig. 3

In the methods used in [32, 33] and [34], one needs: (i) analytical solutions for displacements in the stringer at a given fastener location due to concentrated forces at any other fastener location; (ii) analytical solutions for displacements in the sheet, at a given fastener location, for the Problem B; (iii) analytical solution for displacements, in Problem C, at any given fastener location, due to a set of concentrated forces at any other fastener location; and (iv) analytical solutions for displacements in the sheet, at a given fastener location, for Problem D. It should be noted that not all of these analytical solutions are readily available in the literature; and, when available, they are not suitable for the multiple crack case. When the displacement solutions (i) through (iv) above are used in Equation (94), one obtains the linear system of equations:

$$C \cdot F = A \quad (95)$$

where  $C$  is the "flexibility matrix" as assembled from solutions (i) through (iv) above,  $\underline{F}$  is the vector of fastener forces (at all fastener locations) acting on the sheet, and  $\underline{A}$  is the given vector that accounts for fastener flexibility, etc. In equation (95) the vector  $\underline{F}$  is the unknown.

Once Eq. (95) is solved for the vector  $\underline{F}$ , the problem of the fracture of the sheet alone, i.e., Problem A above, is well-posed [i.e., the far-field hoop stress, as well as the fastener reaction forces, are known]. This Problem A is broken up into Problems B, C, and D as above. The stress singularities at the crack-tips arise only in *Problems B and D*. Thus, the stress-intensity factor at a crack-tip can be written as:

$$K = K_B + K_D \quad (\text{at each crack - tip}) \quad (96)$$

It should be recognized that the key element in the approach described above, is the assembly of the matrix  $C$  in Eq. (95) from the various analytical solutions as described under (i) through (iv) earlier. Thus, the approach is by no means a trivial one for adaptation on a modern personal digital computer. Further, in writing Eq. (96), Problems B and D are assumed to correspond to *infinite* domains in [33] and [34]. Thus, finite-dimension correction factors are ignored, which is not necessarily a conservative approximation.

### 3.1 A SIMPLE FINITE ELEMENT ALTERNATING METHOD FOR STIFFENED PANELS WITH MULTIPLE CRACKS

A straight-forward finite element modeling (including the crack-tip singularities) of a cracked, stiffened, fuselage panel, especially when used for a parametric study during the service/design phase of the aircraft, is prohibitively expensive [32]. In order to circumvent this, displacement compatibility methods wherein the panel is assumed to be infinite, have been developed in [32, 33] and [34], among others.

In the following, a very simple alternative to the current state-of-the-art displacement compatibility approach, is presented. The proposed method essentially involves two steps: (i) evaluation of the fastener reaction forces on the sheet through a very coarse finite element model, wherein the details of the crack-tip singularities are not modeled; and (ii) the application of the finite-element-alternating method, employing the same finite element mesh as in stage (i) for the *unstiffened, uncracked* panel, with the applied membrane hoop stress, and the fastener reaction forces as solved in stage (i), to determine the fracture parameters. The present method may be classified as the now-standard finite element stiffness (or displacement) method.

#### Stage (i) Solution

The skin is discretized using a very coarse finite element mesh, such that the fastener locations are taken to be the nodes of the finite element mesh. The axial deformation of the stiffener is modeled by using the conventional "truss-type" elements. Since the fastener shear forces are usually offset from the stiffener neutral axis, an out-of-plane bending is also induced in the stiffeners. The out-of-plane bending deformation of the stiffener, between two fasteners, is given by the elementary beam theory:

$$\delta_b = C \left( \frac{ML}{E_{St}I} \right) = a_b F \quad (97)$$

where:

$C$  = distance from the neutral axis of the stiffener to the point of action of the fastener shear force

$I$  = stiffener cross-sectional inertia

$L$  = distance between two fasteners, or the length of the stiffener "truss" element

$E$  = Young's modulus of stiffener material

$M = F \cdot C$ ; where  $F$  is the force in the truss element

Note that  $\delta_b$  is in the same direction, as the stiffener axial force,  $F$ . From (97), it is seen that:

$$a_b = \left( \frac{C^2 L}{E_{St}I} \right) \quad (98)$$

The axial stretch of the stringer is given by:

$$\delta_a = \frac{F}{E_{St}A} \equiv (a_S)F ; \quad a_S = \frac{1}{E_{St}A} \quad (99)$$

We use the total axial deformations at the ends of the stringer, at the points where the stringer is attached to the skin, as the generalized degrees of freedom for the stringer. For these degrees of freedom, the stiffness matrix of the stringer element is given by:

$$K_{St}^e = \begin{bmatrix} \frac{1}{(a_b + a_s)} & \frac{-1}{(a_b + a_s)} \\ \frac{-1}{(a_b + a_s)} & \frac{1}{(a_b + a_s)} \end{bmatrix} \quad (100)$$

The flexibility of the fasteners has been found [33] to be an important factor that influences the stress-intensity factors for a crack in the stiffened skin. If  $Q$  is the shear force acting on the fastener, the shear deformation of the fastener can be represented by the empirical relation [33].

$$\delta_F = \frac{1}{E_{Sh}D} \left[ A + C \left( \frac{D}{B_1} + \frac{D}{B_2} \right) \right] F \equiv a_F Q \quad (101)$$

where

$E_{Sh}$  = modulus of sheet material

$D$  = rivet diameter

$B_1$  and  $B_2$  = thicknesses of joined sheets

$A = 5.0$  for Al rivets and 1.66 for steel fasteners

$C = 0.8$  for Al rivets and 0.86 for steel fasteners

The "stiffness" of the rivet in shear is thus given by:

$$K_F = \frac{E_{Sh}D}{\left[ A + C \left( \frac{D}{B_1} + \frac{D}{B_2} \right) \right]} \quad (102)$$

consider, for simplicity (but without any loss of generality), that the skin is discretized into finite elements, with nodes being only at the locations of the fasteners; and likewise, the stringers are discretized into finite elements with nodes being only at the fastener locations. Let the number of fasteners be  $N$ . Let the number of stringer elements be  $N_{St}$ ; and the number of sheet elements be  $N_{Sk}$ . Let the generalized displacements of the skin at the nodes of the finite element mesh be denoted as  $q_{sk}$ ; and those of the stringers at the nodes be denoted by  $q_{st}$ . Then the total strain energy of the stiffened fuselage skin, with flexible fasteners, is given by:

$$W = \sum_{ele=N_{Sk}} \frac{1}{2} q_{Sk}^t K_{Sk}^e q_{Sk}^e + \sum_{N_{St}} \frac{1}{2} q_{St}^t K_{St}^e q_{St}^e + \sum_N \frac{1}{2} K_F (q_{Sk} - q_{St})^2 \quad (103)$$

where  $q_{Sk}^e$  is the vector of nodal displacements of a skin element; and  $q_{St}^e$  is the vector of nodal displacements of a stringer (stiffener) element.

Let  $q_{Sk}$  be the master vector of nodal displacements of the skin;  $K_{Sk}$  the assembled nodal stiffness matrix of the skin;  $q_{St}$  be the master vector of nodal displacements of the stringers; and  $K_{St}$  the assembled nodal



stiffness matrix of the stringers. Let  $K_F$  be the "assembled" (diagonal) stiffness matrix of the fasteners; ie.,

$$K_F = \begin{matrix} K_F & 0 & 0 & 0 & \cdots & 0 \\ 0 & K_F & 0 & 0 & \cdots & 0 \\ 0 & & & & & 0 \\ \vdots & & & & & \vdots \\ 0 & 0 & \cdots & \cdots & K_F & \end{matrix} \quad (104)$$

$N \times N$

Then,  $W$  of Eq. (103) can be written as:

$$W = \frac{1}{2} q_{Sk}^t (K_{Sk} + K_F) q_{Sk} + \frac{1}{2} q_{St}^t (K_{St} + K_F) q_{St} - q_{Sk}^t K_F q_{St} \quad (105)$$

If the fastener flexibility is ignored, then  $q_{Sk} \equiv q_{St}$ ; and equation (105) reduces to:

$$W = \frac{1}{2} q_{Sk}^t (K_{Sk} + K_{St}) q_{Sk} \quad (106)$$

The potential of the external forces (the hoop stress in the fuselage) may be represented, in general, as:

$$U = (q_{Sk}^t Q_{Sk} + q_{St}^t Q_{St}) \quad (107)$$

Let  $\pi = W - U$ . The finite element equations that arise from the vanishing of the variation of  $\delta\pi$  (ie.,  $\delta\pi = 0$ ) are given by:

$$(K_{Sk} + K_F) q_{Sk} - K_F q_{St} = Q_{Sk} \quad (108a)$$

and

$$(K_{St} + K_F) q_{St} - K_F q_{Sk} = Q_{St} \quad (108b)$$

for the case of flexible fasteners. Eqs. (108) may be rearranged as:

$$\begin{matrix} (K_{Sk} + K_F) & (-K_F) & q_{Sk} & Q_{Sk} \end{matrix} \quad (109)$$

$$\begin{matrix} -(K_F) & K_{St} + K_F & q_{St} & Q_{St} \end{matrix}$$

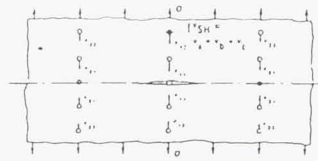
After the imposition of appropriate boundary conditions,  $q_{Sk}$  and  $q_{St}$  can be solved for, from Eq. (109). Once  $q_{Sk}$  and  $q_{St}$  are solved for, the *reactions* of the *stiffeners* on the skin, at the locations of the fasteners, can be easily calculated as:

$$F_{stiffener} = K_F (q_{St} - q_{Sk}) \equiv Q_{St} - K_{St} q_{St} \quad (110)$$

with care being exercised to determine the direction of these reactive forces.

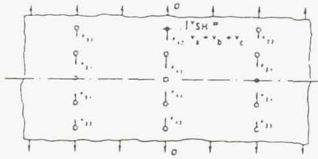
Once the effects of the stringer (with flexible rivets) on the skin are determined, one can consider the free-body diagram of the *cracked skin alone* [See Fig. 4]; the skin being subject to the far-field hoop stresses, and the stringer reaction forces. The stress-intensity factors for the multiple cracks in the skin, subjected to these forces, may now be determined in the stage (ii) alternating solution.

CONSIDER SKIN ONLY:



$F_{ij}$ : Fastener  
forces on sheet.

ASSUME NO CRACK:

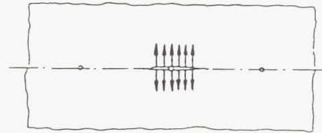


Use coarse  
mesh FEM.

$[K_{sheet}$  is already formed once before.

$$F_{stiff} \text{ is known} \Rightarrow K_{sheet} = F_{stiff} + F_{\sigma}^{(Un-cracked \text{ sheet})}$$

Solve the un-cracked sheet using very coarse  
FEM.



[Pressurized crack only]

Fig. 4

### Stage (ii) solution

In the finite element alternating method to determine the k-factors for multiple cracks in the skin [after the effects of the stiffeners and fasteners are isolated as in stage (i) above], one may employ the same finite element mesh as in stage (i), to model the *un-cracked, unstiffened* free-body of the skin alone. In stage (ii), as explained in Section 2 of this paper, one needs to know: (a) the tractions to be erased at the locations of the cracks in an otherwise un-cracked skin; and (b) the analytical solution for a crack subjected to arbitrary tractions in an infinite body.

### 3.2 AN EXAMPLE PROBLEM ANALYZED BY THE ALTERNATING METHOD

To illustrate the application of the proposed "direct-stiffness" finite element alternating methodology, for analyzing cracked stiffened fuselage panels, the example problem as shown in Fig. 5 has been analyzed. The problem involves a stiffened panel, with a central crack and a broken stiffener, with the crack being symmetrically located with respect to the broken stiffener. For the purpose of illustration of the types of finite element meshes that may be sufficient in the present methodology, the fastener flexibility is ignored in the analysis. A typical finite element mesh that is used is shown in the inset of Fig. 5. The stress-intensity magnification factors for various crack lengths are shown in Fig. 6, which also shows the convergence of the results with mesh refinement. It is seen that the results are insensitive to the finite element mesh size, with acceptable results being obtained for a  $(16 \times 3)$  mesh. It is also seen that the stress-intensity magnification factor decreases as the crack-tip approaches the stiffener.

#### 4 MSD NEAR A ROW OF FASTENER HOLES

A typical multiple site damage near a row of fastener holes is illustrated schematically in Figs. 7 and 8. In example 4.2, the crack was supposed to be much larger in size as compared to the rivet diameter; and thus, the detailed distribution of stresses, on the hole-surface in the skin, due to fastener reaction forces, was not a factor in determining the fracture parameters near the crack-tip. In fact, in Section 3, these reaction forces were simply treated as concentrated forces. However, in a typical MSD situation as depicted in Figs. 7 and 8, the fastener interaction stresses on the hole-surface in the skin are likely to play a significant role in the fracture parameters for the cracks emanating from fastener holes in the skin.

The fastener reaction forces can still be determined for the lap-splice joint configuration of Figs. 7 and 8, through the direct-stiffness finite element method presented above in Section 4. Once these reaction forces, treated as concentrated forces in Section 3, are determined, one may use the known elasticity solution, to approximate the detailed stress-field on the hole-surface in the skin, that is equivalent to these concentrated forces. Under the action of these fastener interaction stress-fields, the stress-intensity factors for MSD near fastener holes can be determined using the alternating method described earlier. Such methods are currently being developed at Georgia Tech.

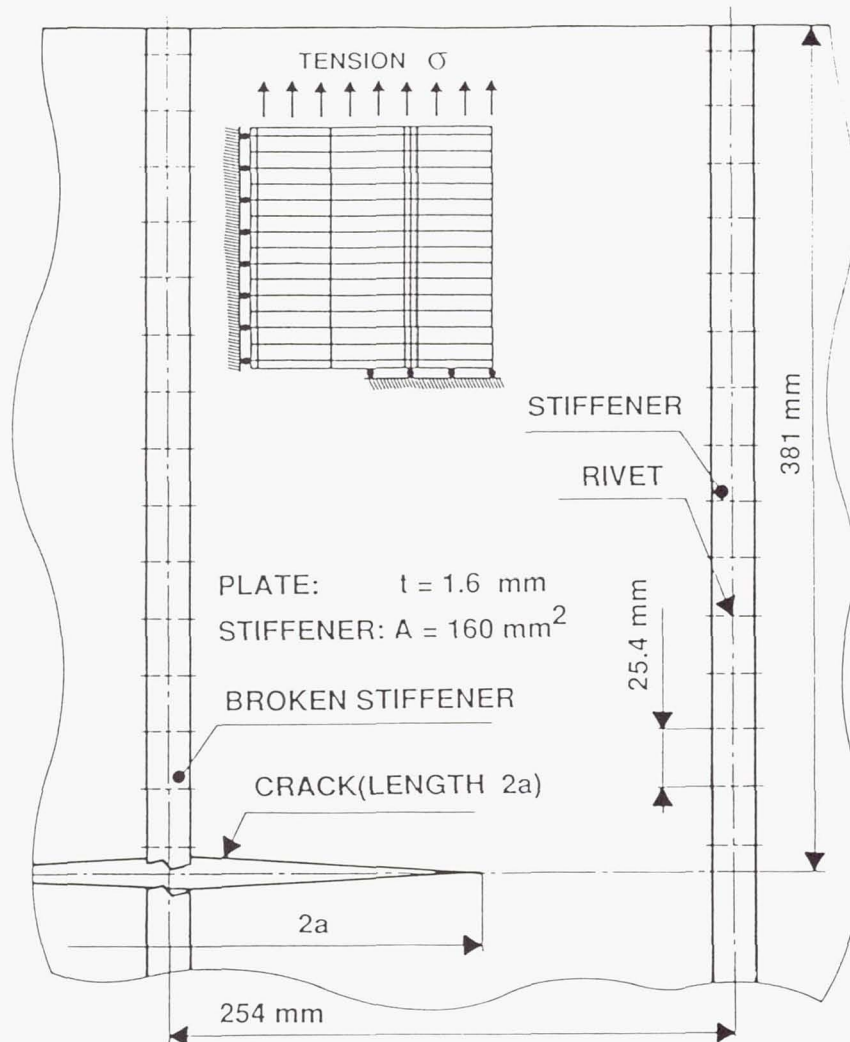


Fig. 5



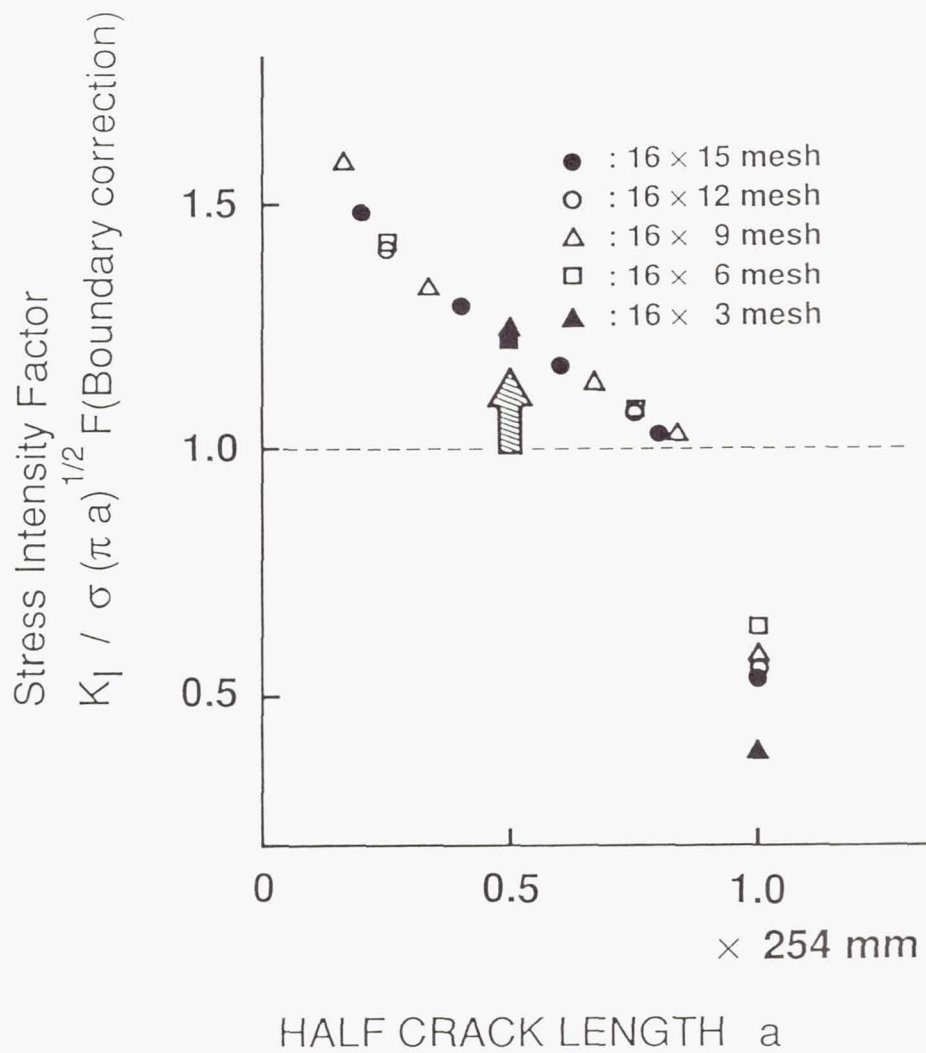


Fig. 6

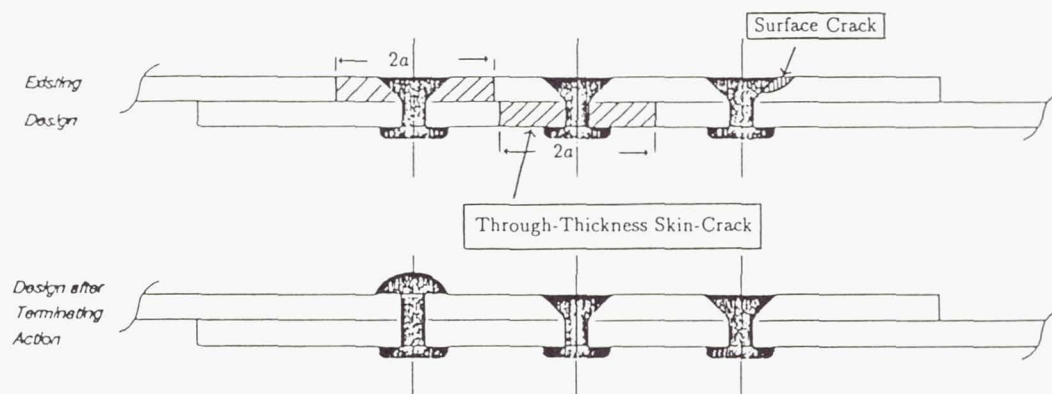
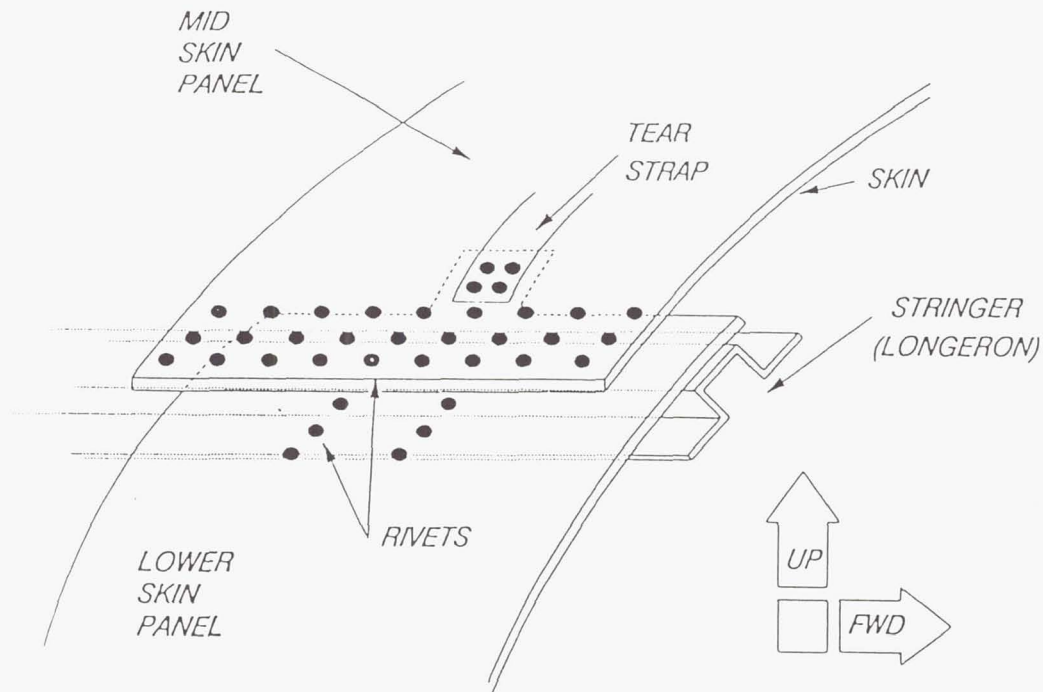


Fig. 7



Sketch of Typical Riveted, Lap-Jointed Fuselage Panels  
Fig. 8

## 5 ANALYSIS OF CRACKS WITH BONDED REPAIR PATCHES

Repair of cracked structures with bonded composite laminate patches appears to be a cost-effective and reliable way of enhancing the fatigue life of fuselage panels [35]. The load-transfer to the composite patch is the primary cause for lessening the stress-intensity near the crack-tip in the main panel, and thus improves its fatigue life. This reduction in the stress-intensity for a crack in the main panel depends on the laminate properties, its thickness, the properties of the adhesive material, and the thickness of the adhesive layer. Analytical solutions have been obtained for: (i) the case of an infinite isotropic material panel with a crack, bonded to an infinite orthotropic panel, and (ii) the case of an infinite isotropic panel with a crack emanating from a hole, bonded to an infinite orthotropic panel. These analytical solutions have been repeated for orthotropic patches of infinite width and finite height (perpendicular to the crack-line in the isotropic panel), with the edges of the patch being parallel to the crack-axes. These analytical solutions, when implemented in the finite-element alternating method, would provide useful design tools for *designing composite patches of arbitrary shape* to arrest further growth of a crack emanating from stress-concentrations in the main panel. Such a design tool, for implementation on a personal computer, is being developed at Georgia Tech.

## 6 WEIGHT-FUNCTIONS FOR 2 AND 3-D ELASTIC CRACK PROBLEMS

The concept of weight functions for elastic crack problems dates back to the work in [36] and [37] see also [38]. The "weight function" may generally be viewed as the appropriately normalized rate of change of displacements (at the surface where tractions are applied, or in the domain where body forces are applied) due to a unit change in the crack length for a *reference state* of loading. The practical importance of the concept of the weight functions lies in the fact that, when the weight functions are evaluated from a (*perhaps simple*) *reference state of loading*, then the stress-intensity factors for any arbitrary state of loading can be computed by using an integral of the worklike product between the applied tractions at a point on the surface in the arbitrary state of loading and the weight function for the reference state at the

same point.

The energy-release due to a unit crack-extension in a cracked elastic body, subject to a system of surface tractions, (We assume, for simplicity, that the surface  $S_u$  where non-zero displacements are prescribed, is zero. *One can easily generalize the ensuing discussion to the situation when  $S_u$  is nonzero.*), and body forces, is given by:

$$\mathcal{G} = \int_{S_t} t_i \frac{du_i}{da} dS + \int_V f_i \frac{du_i}{da} dV - \frac{d}{da} \int_V W dV \quad (111)$$

where  $t_i$  are tractions applied at the surface  $S_t$ ;  $f_i$  are body forces in the domain  $V$ ;  $u_i$  are displacements, and  $W$  is the strain-energy density (internal energy in mechanical work). Eq. (111) may be written as:

$$\mathcal{G} + \int_{S_t} \frac{dt_i}{da} u_i dS + \int_V \frac{df_i}{da} u_i dV = \frac{d}{da} \left\{ \int_{S_t} t_i u_i dS + \int_V f_i u_i dV - \int_V W dV \right\} \quad (112a)$$

or

$$\mathcal{G} da + \int_{S_t} dt_i u_i dS + \int_V df_i u_i dV = -d\pi \quad (112b)$$

Let the reference load state be characterized by a parameter  $\lambda$ . Thus,

$$\mathcal{G} da + d\lambda q(\lambda) = -d\pi \quad (113)$$

where

$$q(\lambda) = \int_{S_t} \hat{t}_i u_i dS + \int_V \hat{f}_i u_i dV \quad (114)$$

where,  $dt_i = d\lambda \hat{t}_i$ ;  $df_i = d\lambda \hat{f}_i$ ; and, in general, in a nonlinear elastic problem, the generalized displacement  $q$  is a nonlinear function of  $\lambda$ . Equation (113) implies that:

$$\left( \frac{\partial \mathcal{G}}{\partial \lambda} \right)_a = \left( \frac{\partial q}{\partial a} \right)_\lambda \quad (115)$$

Consider a linear-elastic homogeneous solid, that is in general *anisotropic*, and consider the case when the crack is at an arbitrary angle to the material directions, and under a general mixed mode loading. The energy release rate,  $\mathcal{G}$ , for a mixed-mode crack in a monoclinic anisotropic solid may be written as:

$$\mathcal{G} = AK_I^2 + BK_{II}^2 + CK_I K_{II} \quad (116)$$

where

$$\begin{aligned} A &= -\frac{\pi}{2} a_{22} \operatorname{Im} \left( \frac{\mu_1 + \mu_2}{\mu_1 \mu_2} \right) \\ B &= \frac{\pi}{2} a_{11} \operatorname{Im}(\mu_1 + \mu_2) \\ C &= \frac{\pi}{2} \left\{ -a_{22} \operatorname{Im} \left( \frac{1}{\mu_1 \mu_2} \right) + a_{11} \operatorname{Im}(\mu_1 \mu_2) \right\} \end{aligned} \quad (117)$$

where  $a_{ij}$  are material constants in the relation  $\varepsilon_i = a_{ij} \sigma_j$  ( $i, j = 6$ ) and  $\mu_j$  are the complex roots of the characteristic equation,  $a_{11} \mu_j^4 - 2a_{16} \mu_j^3 + (2a_{12} + a_{66}) \mu_j^2 - 2a_{26} \mu_j + a_{22} = 0$ . See [31] for further details. In the case of isotropy, (118) reduces to

$$\begin{aligned} A &= \frac{1}{H}; \quad B = \frac{1}{H}; \quad C = 0 \\ H &= E/(1 - \nu^2) \quad \text{plane strain}; \quad H = E \quad \text{plane stress} \end{aligned} \quad (118)$$



We now consider the *simultaneous action* of 2 load-systems on the cracked body. The load system is:

$$(\lambda^1 \hat{t}_i^1 + \lambda^2 \hat{t}_i^2) \text{ at } S_i; \quad \text{and} \quad (\lambda^1 \hat{f}_i^1 + \lambda^2 \hat{f}_i^2) \text{ in } V \quad (119)$$

When (119) is used in (113) one obtains:

$$\mathcal{G} da + d\lambda^R C_{Rn} \lambda^n = -d\pi; \quad R, n = 1, 2 \quad (120)$$

or

$$\frac{\partial \mathcal{G}}{\partial \lambda^R} = \frac{dC_{Rn}}{da} \lambda^n \quad (121)$$

and

$$\begin{aligned} \frac{dC_{Rn}}{da} &= \int_{S_i} \hat{t}_i^n \frac{d\hat{u}_i^R}{da} dS + \int_V \hat{f}_i^n \frac{d\hat{u}_i^R}{da} dV \\ R, n &= 1, 2 \text{ load cases} \quad i = 1, 2, 3 \end{aligned} \quad (122)$$

The  $K$ -factors under the combined mode loading are

$$K_I = \hat{K}_I^m \lambda^m; \quad K_{II} = \hat{K}_{II}^m \lambda^m \quad (123)$$

[sum on  $m = 1, 2$ ] and

$$\mathcal{G} = A \hat{K}_I^m \hat{K}_I^n \lambda^m \lambda^n + B \hat{K}_{II}^m \hat{K}_{II}^n \lambda^m \lambda^n + C \hat{K}_I^m \hat{K}_{II}^n \lambda^m \lambda^n \quad (124)$$

[sum  $m, n = 1, 2$ ]. Using (124) in (121), and observing that the resulting equation is valid for arbitrary  $\lambda^1$  and  $\lambda^2$  one obtains:

$$(2A + C) \hat{K}_I^R \hat{K}_I^n + (2B + C) \hat{K}_{II}^R \hat{K}_{II}^n = \frac{dC_{nR}}{da} \quad (125)$$

$$\begin{aligned} &= \int_{S_i} \hat{t}_i^n \frac{d\hat{u}_i^R}{da} dS + \int_V \hat{f}_i^n \frac{d\hat{u}_i^R}{da} dV \\ R, n &= \text{load cases} \end{aligned} \quad (126)$$

Let  $R$  be a known reference load-state, for which the solution, i.e.,  $\hat{K}_I^R$ ,  $\hat{K}_{II}^R$ , and  $(d\hat{u}_i^R/da)$  are known, and  $n$  is an arbitrary load-state for which the mixed mode factors  $K_I^n$  and  $K_{II}^n$  are to be computed. Eq. (126) is thus a single equation governing the two unknowns  $K_I^n$  and  $K_{II}^n$ . By writing Eq. (126) for *two known and linearly independent reference states*, two equations for two unknowns  $K_I^n$  and  $K_{II}^n$  can be obtained. The solution of these equations can be seen to be:

$$K_I = \frac{\hat{K}_{II}^{R2} \lambda^m}{K^R (2A + C)} \frac{dC_{mR1}}{da} - \frac{\hat{K}_{II}^{R1} \lambda^m}{K^R (2A + C)} \frac{dC_{mR2}}{da} \quad (127)$$

(no sum on  $m$ )

$$K_{II} = \frac{\hat{K}_I^{R1} \lambda^m}{K^R (2B + C)} \frac{dC_{mR2}}{da} - \frac{\hat{K}_I^{R2} \lambda^m}{K^R (2B + C)} \frac{dC_{mR1}}{da} \quad (128)$$

(no sum on  $m$ )

and

$$K^R = \hat{K}_I^{R1} \hat{K}_{II}^{R2} - \hat{K}_I^{R2} \hat{K}_{II}^{R1} \quad (129)$$

where  $(dC_{mR1}/da)$  etc. are defined from (122) by replacing  $R$  by  $R1$  etc. It is seen that for  $K^R$  to be *nonzero*, the reference states should not be both of either Mode I or Mode II. Furthermore, in a general

anisotropic body with an arbitrarily oriented crack, the reference states  $R1$  and  $R2$  can be taken to either loads on external surfaces, or tractions on the crack-faces themselves.

Thus, to evaluate the mixed-mode load factors for any reference state  $m$ , one only needs the appropriately normalized weight-functions,  $(du_i^{R1}/da)$  and  $(du_i^{R2}/da)$ . In the following we discuss some recent work on computational methods for these weight functions, for anisotropic or isotropic materials.

### 6.1 WEIGHT FUNCTIONS, USING FINITE ELEMENT/BOUNDARY ELEMENT MODELS OF ONLY UNCRACKED STRUCTURES

Recently [15, 39] and [40, 41] simple methods for computing weight functions were developed using finite element or boundary element models of only the *uncracked* structure.

It is worth noting that, due to the complications of the fundamental solutions (for a point load) for a general anisotropic medium, the boundary element method is not convenient for application to the anisotropic solids. However, it is well known that the Galerkin finite element method does not have this restriction. The authors in [15, 39] use the finite-element alternating method (for general anisotropic solids) and the authors in [40, 41] use the boundary-element alternating method (for isotropic solids) in computing the weight functions.

While the load-systems are, in general, considered to be at  $S_t$  and in  $V$ , it is convenient to consider only the complementary problem of tractions on the crack-face alone, and consider the case, for simplicity, when the body forces are zero. Thus, hence forth we treat  $S_t$  to be the crack-face alone. It is seen that the weight-functions  $(du_i^R/da)$  on the crack-face will be singular (of the  $r^{-1/2}$  type, where  $r$  is the distance from the crack-tip). Thus, special quadrature rules are needed to integrate the quantity  $t_i^m [du_i^R/da]$  on the crack-face [15, 39].

The following solution procedure is adopted to compute the weight-functions for an embedded or edge crack in a general anisotropic, finite-dimensional structure, when the crack is oriented arbitrarily with respect to the material axes of anisotropy.

(A) Consider two different reference states: one a normal pressure (say constant) on the crack-face and the second a shear traction (say constant) the crack-face. These two load-states are labelled  $R1$  and  $R2$  respectively. Henceforth it is understood that the following steps are carried out for states  $R1$  and  $R2$  respectively.

(B) The start with, treat the problem as one of an *infinite domain*. As discussed in Section 2.3 of this paper, expand the applied tractions on the crack face in the form [30]:

$$\tau_{yy} - i\tau_{xy} = -[p(t) + is(t)] = \sum_{n=1}^N b_n U_{n-1}(t) \quad |t| \leq a \quad (130)$$

where  $U_{n-1}(t)$  is the Chebyshev polynomial of the second kind, defined as:

$$U_n = \sin[(n+1)t]/\sin \theta; \quad t = a \cos \theta \quad (131)$$

and  $b_n$  are the parameters determined by curve-fitting. For this applied loading on the crack-face in an infinite anisotropic body, the solution for the  $K$ -factors, far-field stresses, and crack-face displacements, can be derived [30], as:

$$K_I - iK_{II} = \mp \sqrt{\pi a} \sum_{n=1}^N b_n \quad x = \pm a \quad (132)$$

$$\begin{aligned} u_x &= 2 \operatorname{Re}[p_1 \phi(z_1) + p_2 \psi(z_2)] \\ u_y &= 2 \operatorname{Re}[q_1 \phi(z_1) + q_2 \psi(z_2)] \end{aligned} \quad (133)$$

$$\begin{aligned} \tau_{xx} &= 2 \operatorname{Re}[s_1^2 \phi'(z_1) + s_2^2 \psi'(z_2)] \\ \tau_{yy} &= 2 \operatorname{Re}[\phi'(z_1) + \psi'(z_2)] \\ \tau_{xy} &= -2 \operatorname{Re}[s_1 \phi'(z_1) + s_2 \psi'(z_2)] \end{aligned} \quad (134)$$



$$\begin{aligned}
2\phi(z_1) &= \left( \frac{s_2}{s_2 - s_1} \right) \sum_{n=1}^N \operatorname{Re}(b_n) \frac{R_n(z_1)}{n} + \frac{1}{(s_1 - s_2)} \sum_{n=1}^N i \operatorname{Im}(b_n) \frac{R_n(z_1)}{n} \\
2\psi(z_1) &= \left( \frac{s_1}{s_1 - s_2} \right) \sum_{n=1}^N \operatorname{Re}(b_n) \frac{R_n(z_2)}{n} + \frac{1}{(s_2 - s_1)} \sum_{n=1}^N i \operatorname{Im}(b_n) \frac{R_n(z_2)}{n}
\end{aligned} \tag{135}$$

For further details of the definitions of various parameters in (134), see [30] (note the coordinate system:  $x$  along the crack,  $x = \pm a$  for the crack-tips, and  $y$  normal to the crack).

(C) Compute the  $K$ -factors for step (B) using (132).

(D) Compute the crack-face displacements for step (B) using (133). Note that, once the coefficients  $b_n$  of (130) are known, the crack-face displacements  $u_x$  and  $u_y$  from (135) and (133) are known in an *analytical* form, with their dependence on the crack length being explicitly known (see [30] for details).

(E) Compute the *tractions at the boundaries of the given finite-dimensional structure*, using (134). Call these residual boundary-traction system as  $T$  (recall that steps (B) onwards are repeated for reference systems  $R1$  and  $R2$  of step (A)).

(F) From the analytical expressions for  $u_i$  at the crack-face as determined in step (D), determine the *analytical* expression for  $(du_i/da)$  by differentiating  $u_i$  w.r.t.  $a$ . Note that  $(du_i/da)$  will be infinite at the crack-tip  $x = \pm a$ .

It is important to remember that the present step (F) is still *based on an analytical solution*. No *finite element* or *boundary-element* models, and no *virtual-crack extensions*, and no *finite-difference* methods are used in computing  $(du_i^R/da)$ .

(G) Now consider the *finite element model of the uncracked structure* of the *given geometry*, and anisotropic material [In the case of *isotropic* material, a boundary-element model (with only the boundary being discretized) of the uncracked structure is far more efficient [40, 41]. Apply the reverse of the traction system  $T$  as determined in step (E) above, on the boundaries of the *uncracked structural model*. From the finite-element (or boundary element) solution, find the *tractions at the location of the crack* in the uncracked structure, and label this traction system as  $R_c$ .

(H) Reverse the system  $R_c$  on the crack faces. Go back to step (B), and repeat steps (B), (C), (D), (E), and (F), for this system  $R_c$  on the crack face in an *infinite domain*. Repeat steps (B) to (H) until *convergence is obtained*, i.e., the traction system  $T$  in step (E) is *negligible*.

(J) The weight-functions for a finite-dimensional structure of the given geometry, and given crack-oriented, are obtained by summing up all the values of  $(du_i^{R1}/da)$  [and  $du_i^{R2}/da$ ] obtained in step (F) *for all iterations* until convergence is established.

A number of problems has been solved in [15, 39] and [40, 41] to demonstrate the ease and accuracy of the above procedures.

It is important to note, *the crack is not numerically modelled* at all. The dependence of crack plane displacements on crack length as evaluated in steps (D) and (F), for infinite domains, is explicitly known; and *thus the weight-functions are evaluated in the above alternating method, in an analytical sense without using numerical differentiations*. Furthermore, the above mentioned procedures have been documented in the work of [15, 39] to work very well for anisotropic materials with arbitrarily oriented cracks.

For a two-dimensional anisotropic problem, it is possible to develop a boundary element method for mixed-mode crack analysis, *wherein a straight crack is explicitly included in the formulation and not modeled by boundary elements*, by using the fundamental solutions for an infinite cracked anisotropic plate. This was developed by [42]. This boundary integral equation is:

$$C_{ij}U_j = \int_{\partial\Omega} [u_{ji}^* t_j - t_{ji}^* u_j] dA \tag{136}$$



It should be noted that the crack surface,  $S_c$ , is not a part of  $\partial\Omega$  in (136), as the crack is explicitly accounted for in the fundamental solution. By differentiating (136) w.r.t.  $a$ , one obtains the integral equations for  $(du_j/da)$ :

$$C_{ij}(du_j/da) = \int_{\partial\Omega} [u_{ji}^* \frac{dt_j}{da} - t_{ji}^* \frac{du_j}{da} + \frac{du_{ji}^*}{da} t_j - \frac{dt_{ji}^*}{da} u_j] dA \quad (137)$$

By taking the limit on the left-hand-side to  $\partial\Omega$ , one can solve the *boundary* integral equation for the unknown values of  $(du_j/da)$  and  $(dt_j/da)$  at  $\partial\Omega$ . Once these data at  $\partial\Omega$  is known, Eq. (137) simply becomes an integral relation for the interior values of  $(du_j/da)$ . Since the crack surface is interior to  $\partial\Omega$  in the formulation, the crack-surface weight functions can be determined from (137). Such procedures have been reported, along with some examples, in [43]. An advantage of this procedure is its ability to decouple the vector components of crack tip behavior easily.

## 6.2 WEIGHT-FUNCTIONS BY USING DIRECT FINITE ELEMENT/BOUNDARY ELEMENT MODELING OF THE CRACKED STRUCTURE

The earlier class of modeling used only F.E.M. or B.E.M. models of the *uncracked* structure, while the crack was accounted for in some *analytical* fashion. If the material is nonhomogeneous, or if the crack exists in a complicated structural construction such as a bi-material plate or a stiffened plate, etc., the direct *numerical* modeling of the crack itself is unavoidable, to determine the weight functions. We discuss here some advances made recently in this direction. The discussion is limited to the case of *isotropy*.

Consider the analytic relation

$$\mathcal{G} = \int_{S_i} t_i \frac{du_i}{da} dS + \int_V f_i \frac{du_i}{da} dV - \frac{d}{da} \int_V W dV \quad (138)$$

(wherein the existence of  $S_u$  with nonzero values of prescribed  $u_i$  is ignored, for simplicity). In the context of a finite element method, wherein the *cracked structure is modeled* directly by finite elements, Eq. (138) may be written as:

$$\mathcal{G} = \mathbf{Q} \frac{d\mathbf{q}}{da} - \mathbf{q} \mathbf{K} \frac{d\mathbf{q}}{da} - \frac{1}{2} \mathbf{q} \frac{d\mathbf{K}}{da} \mathbf{q} = -\frac{1}{2} \mathbf{q} \frac{d\mathbf{K}}{da} \mathbf{q} \quad (139)$$

where  $\mathbf{Q}$  is the generalized nodal force vector (due to applied loading at arbitrary  $S_i$  and in  $V$ );  $\mathbf{q}$  is the nodal displacement vector, and  $\mathbf{K}$  the stiffness matrix. Eq. (139) follows from (138), since, at equilibrium,  $\mathbf{K}\mathbf{q} \equiv \mathbf{Q}$ .

When a finite element mesh is used near the crack tip, a small change in crack length, by  $da$ , affects only the elements in a core immediately surrounding the crack tip. This is the basic idea behind the stiffness derivative method [44 and 45]. Let the *small domain near the crack tip* be  $V_\varepsilon$ . Thus;

$$\mathcal{G} = -\frac{1}{2} \mathbf{q}_\varepsilon \frac{d\mathbf{K}_\varepsilon}{da} \mathbf{q}_\varepsilon \quad (140)$$

where  $(\ )_\varepsilon$  indicates the quantity  $(\ )$  in the region  $V_\varepsilon$ . In (140),  $(d\mathbf{K}_\varepsilon/da)$  is determined by the finite difference relation:

$$\frac{\mathbf{K}_\varepsilon(a + \Delta a) - \mathbf{K}_\varepsilon(a)}{\Delta a}$$

Eq. (140) can be applied to the *reference state*, to determine the  $K$ -factors for the reference state. However, if the reference state is of mixed mode loading, for isotropic materials,  $\mathcal{G} = (K_I^2 + K_{II}^2)/H$ , and a mode separation is necessary. Thus within the core-region  $V_\varepsilon$  [which is certainly much smaller than the cracked structure], one may decompose  $\mathbf{q}^\varepsilon$  into mode I and mode II parts; by using the relations:

$$\begin{Bmatrix} u_1 \\ u_2 \end{Bmatrix} = \begin{Bmatrix} u_1^I \\ u_2^I \end{Bmatrix} + \begin{Bmatrix} u_1^{II} \\ u_2^{II} \end{Bmatrix} = \frac{1}{2} \begin{Bmatrix} u_1 + u_1' \\ u_2 - u_2' \end{Bmatrix} + \frac{1}{2} \begin{Bmatrix} u_1 - u_1' \\ u_2 + u_2' \end{Bmatrix} \quad (141)$$

where 1 and 2 are directions along and normal to the crack axis, respectively, and where ( ) denotes a quantity at a point  $p$  in the upper portion of the cracked plate, and ( )' is the respective quantity at a point  $p'$  which is the mirror image in the crack axis of  $p$ . Thus  $q_\varepsilon = q_\varepsilon^I + q_\varepsilon^{II}$ .

Thus

$$\frac{(K_I^R)^2}{H} = -\frac{1}{2} q_\varepsilon^I \frac{dK_\varepsilon}{da} q_\varepsilon^I \quad (142)$$

and

$$\frac{(K_{II}^R)^2}{H} = -\frac{1}{2} q_\varepsilon^{II} \frac{dK_\varepsilon}{da} q_\varepsilon^{II} \quad (143)$$

where  $q_\varepsilon^I$  is the vector of appropriate nodal displacements  $u_i^I$ , etc. Now we consider the problem of determining the weight functions for the reference state of mixed mode loading. To this end consider the finite element equilibrium equation for the entire *cracked structure* loaded under mixed mode reference load:

$$Kq = f \quad (144)$$

For a fixed-loading, the weight-functions *everywhere in the structure* can be derived, from (154), as:

$$K \frac{dq}{da} = -\frac{dK_\varepsilon}{da} q \equiv \frac{dK_\varepsilon}{da} q_\varepsilon \quad (145)$$

From a solution of (145),  $(\frac{du_i}{da})$  is determined for the *reference state* (which can, in general, be a mixed-mode loading), at all nodes at  $S_t$ , and in  $V$  (including the crack-face). The above method, for a pure mode I problem was presented by [46].

Now, we present a simple extension of the "stiffness-derivative" weight-function evaluation method for *mixed-mode* problems. In as much as the stiffness matrix ( $K$ ) and its derivative ( $dK/da$ ) are evaluated once and for all for the structure, Eq. (144) can be solved for *two reference* states, at least one of them mixed-mode,  $R1$  and  $R2$ ; Eqs. (142) and (143) can be solved for mixed-mode  $K$ -factors for the two reference states, i.e.,  $K_I^{R1}$ ,  $K_{II}^{R1}$ ,  $K_I^{R2}$ , and  $K_{II}^{R2}$ . Likewise, Eq. (145) can be solved for the two different reference states, to determine  $(du_i^{R1}/da)$  and  $(du_i^{R2}/da)$  *everywhere in the structure* (i.e. at  $S_t$ , in  $V$ , and on the crack-face) as desired. *For any other arbitrary state of mixed-mode loading*, Eqs. (127) and (128) may be used for determining the mixed-mode  $K$ -factors. Note that this simple procedure leads to *weight-functions everywhere in the structure* (external surfaces, crack faces, and within the body) as may be desired.

On the other hand, the authors in [47], instead of using the procedure as discussed in the above paragraph for mixed-mode problems, proceed to consider only pure-mode I and pure-mode II weight functions,  $(du_i^I/da)$  and  $(du_i^{II}/da)$ , by decomposing the displacement *everywhere in the cracked structure* (not only in  $V_\varepsilon$ ) using Eq. (141). If the weight-functions are sought at the external boundary, pairing the points on the external boundary, and their *mirror images* is geometrically impossible, for *arbitrary-shaped* structures, with arbitrarily oriented cracks. They [47] consider the following equations:

$$K \frac{dq^I}{da} = -\frac{dK_\varepsilon}{da} q^I \quad (146)$$

and

$$K \frac{dq^{II}}{da} = -\frac{dK_\varepsilon}{da} q^{II} \quad (147)$$

Note that  $q^I$  ( $q^{II}$ ) is simply a vector of appropriate displacements  $u_i^I$  ( $u_i^{II}$ ) at each node in the structure, which may not always be geometrically feasible.

Another method for weight-functions which obviates the need for a finite difference evaluation of  $(dK_\varepsilon/da)$  is based on the equivalent domain-integral method for evaluating the energy-release rate. This method

has recently been developed in [48] and [15, 39]. As discussed in [49, 50], the energy-release rate in a 2-D elastic problem can be written (as the equivalent domain integral representation of the  $J$ -integral), as:

$$\mathcal{G} = -\frac{1}{F} \int_{V_s} \left[ W \frac{ds}{dx_1} - \sigma_{ij} \frac{\partial u_i}{\partial x_1} \frac{\partial s}{\partial x_j} \right] dV \quad (148)$$

where  $V_s$  is any arbitrary region near the crack-tip [ $V_s$  is much smaller than the total region  $V$ ];  $s$  is any arbitrary but continuous function which is equal to 1 at the crack-tip, and goes to zero at the boundary of  $V_s$ ;  $F = 1$  in two dimensional crack problems;  $W$  is the stress-work density, and  $u_i$  are displacements.

Suppose that the region  $V_s$  in (148) is taken to be the same as the region  $V_\varepsilon$  considered in Eq. (140) [Even otherwise, if  $V_\varepsilon$  is smaller than  $V_s$ ; since  $(dK/da)$  may be taken to be zero in the region  $V_s - V_\varepsilon$ ; one may rewrite (140) as  $\mathcal{G} = -\frac{1}{2} \mathbf{q}_s (d\mathbf{K}_s/da) \mathbf{q}_s$ , without loss of generality]. Suppose that one introduces a finite element interpolation:

$$\begin{aligned} u_i &= N^k u_i^k & K &= 1, \dots, N \text{ nodes} \\ &= N \mathbf{q} & i &= 1, 2 \end{aligned} \quad (149)$$

Then,

$$W = \frac{1}{2} \sigma_{ij} \varepsilon_{ij} = \frac{1}{2} \sigma_{ij} N_{,j}^k u_i^k \quad (150)$$

Also, we introduce the finite element interpolations,

$$s = N^k s^k \quad (151)$$

Using (149–151) in (148), one has:

$$\begin{aligned} \mathcal{G} &= -\frac{1}{F} \int_{V_s} \left[ \frac{1}{2} N_{,j}^k N_{,1}^L - N_{,1}^k N_{,j}^L \right] \sigma_{ij} u_i^k s^L dV \\ &\equiv -\frac{1}{2} T_i^k u_i^k & k &= 1, \dots, N \text{ nodes} \quad i = 1, 2 \end{aligned} \quad (152)$$

$$\equiv -\frac{1}{2} \mathbf{Q}_s^{*t} \cdot \mathbf{q}_s \quad (153)$$

where the definition of  $\mathbf{Q}_s^*$  is apparent. Thus, when  $V_s \equiv V_\varepsilon$ ; comparing (153) with (140), one has:

$$\frac{d\mathbf{K}_\varepsilon}{da} \mathbf{q}_\varepsilon \equiv \mathbf{Q}_\varepsilon^* \quad (154)$$

Note that  $\mathbf{Q}_\varepsilon^*$  is computed from a simple integral over  $V_\varepsilon$  as in (152a) and (153). Eq. (154) shows that a finite-difference evaluation of  $(d\mathbf{K}_\varepsilon/da)$  as in Eq. (140) can be avoided if the identity in (154) is used, and the energy-release-rate can be computed using (153).

For a fixed reference loading (which can in general be of the mixed-mode type), the weight functions everywhere in the structure (including at the external boundary,  $S_t$ , the crack-face, or in  $V$ ), can now be obtained, using (145):

$$\mathbf{K} \frac{d\mathbf{q}}{da} \equiv -\frac{d\mathbf{K}_\varepsilon}{da} \mathbf{q}_\varepsilon \equiv \mathbf{Q}_\varepsilon^* \quad (155)$$

where  $\mathbf{Q}_\varepsilon^*$  is computed from the domain-integral over  $V_\varepsilon$  as apparent from (153).

Equation (155) is solved for two arbitrary reference states [which are both not either of Mode I or of Mode II type, with loading being either on the external boundary, or on the crack-face] to find  $(du_i^{R1}/da)$  and  $(du_i^{R2}/da)$  that are required in Eqs. (123) and (124) in order to compute, the mixed-mode  $K$ -factors for any other given arbitrary load-state. Note that in (155),  $\mathbf{K}$  is computed only once; and  $\mathbf{Q}_\varepsilon^*$  is computed



separately for each *reference state*. However, examining (152a) and (153) it is seen that the only quantity that is different in integral for  $Q_\varepsilon^*$  in the two *reference states* is  $\sigma_{ij}$  in  $V_\varepsilon$ .

In order to use (127) and (128) to compute the mixed-mode  $K$ -factors for any given arbitrary load-state is the only additional informations needed are the mixed-mode  $K$ -factors ( $K_I^{R1}$ ,  $K_I^{R2}$ ), and ( $K_{II}^{R1}$ ,  $K_{II}^{R2}$ ) for the two *reference states*. For isotropic materials, the mode-decomposition of the energy-release rate of Eq. (152a) can be accomplished by decomposing the displacement, strain, and stress fields in the *core region*  $V_\varepsilon$  near the crack tip [ $V_s$  is much smaller than  $V$ , the total domain]. The displacement decomposition is already given in (151), while the stress decomposition can be written as:

$$\begin{Bmatrix} \sigma_{11} \\ \sigma_{22} \\ \sigma_{12} \end{Bmatrix} = \begin{Bmatrix} \sigma_{11}^I \\ \sigma_{22}^I \\ \sigma_{12}^I \end{Bmatrix} + \begin{Bmatrix} \sigma_{11}^{II} \\ \sigma_{22}^{II} \\ \sigma_{12}^{II} \end{Bmatrix} \equiv \frac{1}{2} \begin{Bmatrix} \sigma_{11} + \sigma'_{11} \\ \sigma_{22} + \sigma'_{22} \\ \sigma_{12} - \sigma'_{12} \end{Bmatrix} + \frac{1}{2} \begin{Bmatrix} \sigma_{11} - \sigma'_{11} \\ \sigma_{22} - \sigma'_{22} \\ \sigma_{12} + \sigma'_{12} \end{Bmatrix} \quad (156)$$

where, ( ) and ( )' are quantities at a point  $p$  in the "upper side" of the crack, and at a point  $p'$  which is a mirror image of  $p$  in the crack-plane. Thus, for a reference state, the individual nodal intensities are computed from:

$$\frac{(K_I^R)^2}{H} = -\frac{1}{F} \int_V \left[ W^I \frac{\partial s}{\partial x_1} - \sigma_{ij}^I \frac{\partial u_i^I}{\partial x_1} \frac{\partial s}{\partial x_j} \right] dV \quad (157)$$

and

$$\frac{(K_{II}^R)^2}{H} = -\frac{1}{F} \int_V \left[ W^{II} \frac{\partial s}{\partial x_1} - \sigma_{ij}^{II} \frac{\partial u_i^{II}}{\partial x_1} \frac{\partial s}{\partial x_j} \right] dV \quad (158)$$

where

$$W^I = \frac{1}{2} \sigma_{ij}^I u_{ij}^I; \quad W^{II} = \frac{1}{2} \sigma_{ij}^{II} u_{ij}^{II} \quad (159)$$

This completes the algorithm for determining the mixed mode  $K$ -factors for an arbitrary given loading, using the weight functions for reference states, derived from the *equivalent domain integral method*, which avoids the need for a finite-difference evaluation of the stiffness derivative ( $dK/da$ ) as in the approaches for Mode I problem given in [46].

Special variational technique for determining directly the weight functions that are singular in the vicinity of the crack-tip (crack front) (and hence have unbounded strain-energy) has been presented in [51]. This variational technique handles both traction and mixed boundary conditions. A finite element implementation of the variational principle has also been given in [51] and this leads to a unified approach in the direct finite element computation of weight functions for all three fracture modes. The authors in [52] have presented weight functions for a semi-infinite crack in a full space of arbitrary anisotropy; in particular, the results for monoclinic solids are presented in closed form. Using such a solution, they [52] applied the variational technique in [51] to determining weight functions for homogeneous and piecewise homogeneous anisotropic (where crack tips do not terminate at the material interface) bodies. Employing the weight functions obtained, they also evaluated the stress intensity factors of a matrix crack in an idealized model of a fiber-reinforced composite laminate under curing conditions. The authors in [53] have recently introduced antiplane strain weight functions for an interface notch in an isotropic bi-crystal and in [52] the same variational technique to determine these notch-interface weight functions was used. Generalizing the weight function concepts [36], the authors in [54] have developed higher order weight functions for calculating power expansion coefficients of an elastic field in a two-dimensional body in the absence of body forces. Integration formulas for the expansion coefficient, in analogy to those for stress intensity factors, are given for interior points and crack tips. Some of these expansion coefficients at an interior point can be related to the image force of a discrete dislocation and those for the crack tip correspond to important fracture parameters as discussed in [54].

### 6.3 WEIGHT FUNCTIONS AND INFLUENCE FUNCTIONS FOR 3-D CRACK PROBLEMS

As aptly noted in [55], "since fracture and fatigue are demonstrably three-dimensional, the technology base needed to describe these processes must be of the same dimensionality if only to describe these events, let

alone predict them". Thus, simple numerical methods of engineering interest are in need of development for 3-dimensional problems. Along with the domain integral methods, the weight function approaches to 3-D problems present interesting possibilities in this regard.

Here, we consider, for simplicity, only mode I crack problems in 3-D isotropic elastic solids. For isotropic solids, containing cracks of arbitrary shape, under mode I loading, the author in [37] has derived the counterpart of Eqs. (127) and (128) [for mode I], as follows:

$$\int_{\Gamma_c} \frac{2}{H} [K_I K_I^R \delta \ell] d\Gamma = \int_{S_t} t_i \delta \ell u_i^R dS + \int_V f_i \delta \ell u_i^R dV \quad (160)$$

where  $\Gamma_c$  is the crack-front;  $K_I$  is the stress-intensity factor (which varies along  $\Gamma_c$ ) for any arbitrary loading  $t_i$  at  $S_t$  and  $f_i$  in  $V$ ;  $K_I^R$  is the stress-intensity factor (which varies along  $\Gamma_c$ ) for the reference loading  $t_i^R$  at  $S_t$  and  $f_i^R$  in  $V$ ;  $S_t$  is the loaded-surface (which may be taken to be the crack-face, without loss of generality);  $\delta \ell$  is a smooth function along  $d\Gamma$  denoting the infinitesimal advance of the crack in a direction locally normal to  $\Gamma$ ;  $\delta \ell u_i^R$  will denote the *first-order variation* in  $u_i^R$  to a change in the crack-front i.e.,  $\delta \ell u_i^R$  is a function of the location in  $S_t$  and  $V$ , and  $H$  is a material constant.

The author in [56] has presented results for the first order variation of an elastic displacement field associated with the arbitrary incremental planar advance of the location of the front of a *half-plane crack* in a loaded elastic full space, and also discussed the relation of such results to a 3-D weight function theory, and derived an expression for the distribution of the mode-I  $K$ -factor for a slightly curved crack-front. Later this work was extended [57] to the mixed mode case.

Recently analytical results for 3-D weight functions were presented [58], for a penny-shaped and a half-plane crack in an elastic full-space, under mixed-mode conditions. Employing these results, and the variational technique in [51], the authors in [54] have determined the Mode I weight functions for both penny-shaped and elliptical cracks in finite bodies.

Here, our objective is to discuss crack-surface weight functions for embedded or surface cracks of the elliptical geometry, i.e., *we treat  $S_t$  to be the crack-face* in (160) and ignore the body forces. For surface or corner flaws of semi- or quarter-elliptical geometry respectively, engineering theories of fatigue crack-growth are often based on the consideration of the  $K$ -factors at the major and minor axis locations ( $x = a$ , and  $y = b$ ), respectively. Thus, one often thinks of a "two-parameter" characterization of the  $K$ -factor variation along the crack-front, for the given arbitrary loading. There are two alternative approaches for the above "two-parameter" characterization. One is directly in terms of the "local" values (or values at major and minor axis points on the ellipse)  $K_I^*$  and  $K_I^{**}$ ; and the other is in terms of "local weighted average" values along specified portions of the crack front,  $\bar{K}_I^*$  and  $\bar{K}_I^{**}$ , defined as:

$$\begin{aligned} (\bar{K}_I^*)^2 &= \frac{1}{\delta A_1} \int_{\Gamma_c} K_I^2 \delta \ell d\Gamma \\ (\bar{K}_I^{**})^2 &= \frac{1}{\delta A_2} \int_{\Gamma_c} K_I^2 \delta \ell d\Gamma \end{aligned} \quad (161)$$

where  $K_I$  and  $\delta \ell$  are as in Eq. (160), and  $\delta A_1$  and  $\delta A_2$  are changes in crack area due to a virtual change in the length of the major and minor axes, respectively. The weight functions for these weighted average values,  $\bar{K}_I^*$  and  $\bar{K}_I^{**}$  have been defined in [59], and used for residual life estimations of complex structures in [59] and [60]. In the following we sketch a procedure for determining the weight functions directly for  $K_I^*$  and  $K_I^{**}$ .

Suppose for the *reference load state*, the  $K$ -factor variation,  $K_I^R(\Gamma)$  is determined from the finite element alternating technique as discussed in Section 3 of this paper. Thus, an analytical expression for  $K_I^R$  as a function of the crack-front coordinate  $\phi$ , is known, wherein the appropriate coefficients are determined from the finite element alternating method. We consider the case of the given loading, and introduce a *trial solution* for  $K_I$  for this loading, in terms of its values,  $K_I$  at  $x = a$  (denoted as  $K_I^*$ ) and  $K_I$  at  $y = b$ , denoted as  $K_I^{**}$ .



In as much as  $\delta\ell$  in Eq. (160) is arbitrary, we introduce two such trial variations: (i) one wherein the major axis is extended by  $(da)$  such that the equation of the ellipse is  $(x/(a+da))^2 + (y/b)^2 = 1$ ; and (ii) the other wherein the ellipse is  $(x/a)^2 + (y/(b+db))^2 = 1$ . Thus,  $\delta\ell$  is given in terms of  $(da)$  or  $(db)$  and the elliptical angle. Also, as discussed in Section 3, the crack surface displacements for the reference load state are known in the form of analytical expressions wherein the coefficients are determined from the finite element alternating method. Thus, for  $\delta\ell$  as in case (i) and (ii), the first-order variations in  $u_i^R$  at  $S_i$  can be determined from analytical expressions where coefficients are numerically determined in the alternating method. Thus, when the known expressions for  $K_I^R(\Gamma)$ ,  $\delta\ell(\Gamma)$ ; and  $\delta_\ell u_i^{(R)}$  are used, and the two-parameter trial function for  $K_I$  (for the given state) are used in Eq. (160), one obtains 2 algebraic equations, (one for each of the two cases of  $\delta\ell$  listed above) governing the two unknowns  $K_I^*$  and  $K_{II}^{**}$ . Thus, one obtains a *weight-function representation* for the *stress-intensity factors at the major and minor axes* of the elliptical (or part-elliptical) flaw under the given state of mode I loading.

Using the VNA analytical solution for an embedded elliptical crack in an infinite body as detailed in Section 2 of this paper, an analytical expression was derived [61] for the first order variation of the crack-face displacement field due to a geometrical perturbation in the major and minor axes of an elliptical crack. Using this, the above described weight-function representation was developed [61] for the  $K$ -factors at the major and minor axes of the (part)-elliptical crack, in the forementioned 2 parameter characterization.

If  $K_I$  variation under the action of an arbitrary crack-face traction is needed all along the crack-front, the *influence function* concept is more useful. For a given surface flaw in given structural geometry, one can generate a stress-intensity factor variation all along the crack-front, for a *given polynomial loading* on the crack-face, using the alternating method described in Section 3. If the given arbitrary crack-face pressure is then decomposed into individual polynomial variations multiplied by an appropriate constant, then the  $K$ -factor variation for the given load can easily be determined by a (weighted) linear superposition of the various influence functions. This has been done for several problems [18, 21].

In general, the advantage of the weight function concept over that of the influence functions is that the former can be used for localized crack-face forces, including point forces, while the latter is more suited for more distributed crack-face tractions of the polynomial type.

## 7 DOMAIN INTEGRAL METHODS FOR COMPUTING FRACTURE PARAMETERS IN 3-DIMENSIONAL CRACK PROBLEMS, UNDER ARBITRARY HISTORIES OF LOADING

It is well-known that energetic methods, such as based on the  $J$ -integral and other crack-tip integral parameters, play an important role in elastic-plastic and inelastic fracture mechanics, under arbitrary histories of loading (see, for instance, [1]). For elastic crack problems, in three-dimensions, the evaluation of the  $J$ -integral (in Mode I problems), the stiffness derivative method [44, 62] and the virtual crack extension method [45] proved to be quite useful. Various extensions to, and a certain variety of improvements of, these methods were recently presented in [63]; [64]; [65]; [66]; [49, 50, 67]. These later methods are currently labeled as the "Domain Integral Methods" for the computation of crack-tip integral parameters in non-elastic fracture under arbitrary load histories. In the following, a brief description of these domain-integral methods, as applicable to the analysis of mixed-mode behaviour of arbitrary shaped cracks (surfaces of discontinuity) in 3-D structures, with elastic-plastic or inelastic material behaviour, and under arbitrary loading, is given.

For *elastic* problems, the energy release rate per unit crack-extension (in a self-similar fashion) is given by:

$$J_1 = \mathcal{G} = \int_{S_i} t_i \frac{du_i}{da} dS + \int_V f_i \frac{du_i}{da} dV - \frac{d}{da} \int_V W dV \quad (162)$$

(assuming that  $S_u$ , where non-zero  $u_i$  are prescribed, is zero). Eq. (162) is valid for mixed-mode loadings, and in 3-D problems, it is understood that  $(du_i/da)$  represents a first-order change in  $u_i$  due to a *local* perturbation in the crack-front, and  $J_1$  is the *local* energy-release rate. In a finite element model, Eq. (162) may be written as:

$$J_1 = Q \frac{dq}{da} - \frac{d}{da} \left( \frac{1}{2} q K q \right) \quad (163)$$



$$= -\frac{1}{2} \mathbf{q} \frac{d\mathbf{K}}{da} \mathbf{q} \quad (164)$$

$$= -\frac{1}{2} \mathbf{q}_o \frac{d\mathbf{K}_o}{da} \mathbf{q}_o \quad (165)$$

Eq. (164) follows from (163) since  $\mathbf{Q} = \mathbf{q}\mathbf{K}$  at equilibrium and Eq. (165) follows from (164) since a change in crack length can be seen to affect the stiffness matrix of only a small-core of material, in the domain  $V_o$ , near the crack front. The evaluation of  $d\mathbf{K}_o/da$  is usually accomplished through a finite difference method [44, 45]. The definitions of  $J_2$  and  $J_3$ , which involve the more mathematical concepts of translation of the crack, in  $x_2$  and  $x_3$  directions [1], are not, in general, amenable to calculation through the above stiffness-derivative methods.

The basic advantages of the domain-integral method can be seen from the following, to be: (i) they allow simple computation of all the 3 components  $J_k$  of the vector  $J$  integral; and (ii) in the case of  $J_1$ , the need for a finite difference evaluation of  $(d\mathbf{K}_o/da)$  is obviated.

In a general 3-D problem, for arbitrary material behaviour and loading, one may define the  $J$ -integral vector components as:

$$J_k = \lim_{\epsilon \rightarrow 0} \int_{\Gamma_\epsilon} \left[ W n_k - \sigma_{ij} \frac{\partial u_i}{\partial x_k} n_j \right] d\Gamma \quad (166)$$

where  $W$  is the density of *stress-work* in arbitrary loading and arbitrary material behaviour;  $\sigma_{ij}$  are stresses,  $u_i$  are displacements, and  $n_k$  are components of the unit normal vector to the surface of the tube at points on contour  $\Gamma_\epsilon$ . In principle it is possible to define  $J_k$  in any coordinate system, but for the purposes of prediction of crack behaviour, it is more convenient to have a local crack-front coordinate system  $x_1, x_2, x_3$ :  $x_1$  is normal to the crack front and lies in the plane of the crack surface,  $x_2$  is orthogonal to  $x_1$  and the crack surface, and  $x_3$  is tangential to the crack-front and in the crack-plane. We introduce the equivalent definition of the near-tip  $J$ -integral along the surface of the tube, as:

$$J_k \Delta = \lim_{\substack{\epsilon/\Delta \rightarrow 0 \\ \Delta \rightarrow 0}} \int_{A_\epsilon} \left[ W n_k - \sigma_{ij} \frac{\partial u_i}{\partial x_k} n_j \right] dA, \quad k = 1, 2 \quad (167)$$

where  $A_\epsilon$  is the surface of a cylinder, with the centerline along the crack-front, its radius being  $\epsilon$ , and the length of the generator being  $\Delta$  along the crack-front. This definition is more convenient for numerical applications. Likewise, one may define the energy-release components for symmetric and anti-symmetric deformation modes, as:

$$G_I \Delta = \lim_{\substack{\epsilon/\Delta \rightarrow 0 \\ \Delta \rightarrow 0}} \int_{A_\epsilon} \left( W^I n_1 - \sigma_{ij}^I \frac{\partial u_i^I}{\partial x_1} n_j \right) dA \quad (168)$$

$$G_{II} \Delta = \lim_{\substack{\epsilon/\Delta \rightarrow 0 \\ \Delta \rightarrow 0}} \int_{A_\epsilon} \left( W^{II} n_1 - \sigma_{ij}^{II} \frac{\partial u_i^{II}}{\partial x_1} n_j \right) dA \quad (169)$$

and

$$G_{III} \Delta = \lim_{\substack{\epsilon/\Delta \rightarrow 0 \\ \Delta \rightarrow 0}} \int_{A_\epsilon} \left( W^{III} n_1 - \sigma_{3j} \frac{\partial u_3}{\partial x_1} n_j \right) dA \quad (170)$$

where the deformation field is decomposed, locally near each differential segment of the crack-front, into symmetrical and skew-symmetrical parts about the crack-plane locally, as follows:

$$\begin{aligned} \{u\} &= \{u^I\} + \{u^{II}\} + \{u^{III}\} \\ &= \frac{1}{2} \begin{Bmatrix} u_1 + u'_1 \\ u_2 - u'_2 \\ u_3 + u'_3 \end{Bmatrix} + \frac{1}{2} \begin{Bmatrix} u_1 - u'_1 \\ u_2 + u'_2 \\ 0 \end{Bmatrix} + \frac{1}{2} \begin{Bmatrix} 0 \\ 0 \\ u_3 - u'_3 \end{Bmatrix} \end{aligned} \quad (171)$$

$$\begin{aligned}
\{\sigma\} &= \{\sigma^I\} + \{\sigma^{II}\} + \{\sigma^{III}\} \\
&= \frac{1}{2} \begin{bmatrix} \sigma_{11} + \sigma'_{11} \\ \sigma_{22} + \sigma'_{22} \\ \sigma_{33} + \sigma'_{33} \\ \sigma_{12} - \sigma'_{12} \\ \sigma_{23} - \sigma'_{23} \\ \sigma_{31} - \sigma'_{31} \end{bmatrix} + \frac{1}{2} \begin{bmatrix} \sigma_{11} - \sigma'_{11} \\ \sigma_{22} - \sigma'_{22} \\ 0 \\ \sigma_{12} + \sigma'_{12} \\ 0 \\ 0 \end{bmatrix} + \frac{1}{2} \begin{bmatrix} 0 \\ 0 \\ \sigma_{33} - \sigma'_{33} \\ 0 \\ \sigma_{23} + \sigma'_{23} \\ \sigma_{31} + \sigma'_{31} \end{bmatrix}
\end{aligned} \quad (172)$$

$$u'_i(x_1, x_2, x_3) = u_i(x_1, -x_2, x_3) \quad (173)$$

$$\sigma'_{ij}(x_1, x_2, x_3) = \sigma_{ij}(x_1, -x_2, x_3) \quad (174)$$

## 7.1 TRANSFORMATION OF DISPLACEMENTS, STRAINS AND STRESSES TO THE CRACK FRONT COORDINATE SYSTEM

We can simplify many developments if this transformation is performed prior to the calculation of  $J$ - and  $G$  components. Let  $X_1, X_2, X_3$  be a global Cartesian coordinate system; and  $x_1, x_2, x_3$  be the crack front coordinate system for a particular point along the crack front. For the definition of a crack front coordinate system at any point, it is sufficient to have the direction cosines for a unit vector along  $x_1$

$$X_p = \{X_{p1}, X_{p2}, X_{p3}\} \quad (175)$$

and for a unit vector along  $x_3$

$$Z_p = \{Z_{p1}, Z_{p2}, Z_{p3}\} \quad (176)$$

Then it is easy to define the orientation of  $x_2$  as

$$Y_p = Z_p X X_p \quad (177)$$

$$Y_{p1} = Z_{p2} X_{p3} - Z_{p3} X_{p2}$$

$$Y_{p2} = Z_{p3} X_{p1} - Z_{p1} X_{p3}$$

$$Y_{p3} = X_{p1} X_{p2} - Z_{p2} X_{p1} \quad (178)$$

We define the coefficients of a transformation matrix  $a_{ij}$  as:

$$\begin{aligned}
a_{11} &= X_{p1} & a_{12} &= X_{p2} & a_{13} &= X_{p3} \\
a_{21} &= Y_{p1} & a_{22} &= Y_{p2} & a_{23} &= Y_{p3} \\
a_{31} &= Z_{p1} & a_{32} &= Z_{p2} & a_{33} &= Z_{p3}
\end{aligned} \quad (179)$$

The transformation of coordinates, displacements, strains, and stresses can be done as follows:

$$x_i = a_{ij} X_j \quad (180)$$

$$u_i = a_{ij} u_i^g \quad (181)$$

$$\varepsilon_{ij} = a_{ip} a_{jq} \varepsilon_{pq}^g \quad (182)$$

$$\sigma_{ij} = a_{ip} a_{jq} \sigma_{pq}^g \quad (183)$$

Here the superscript  $g$  stands for the values in global coordinate system.

## 7.2 EDI-TECHNIQUE FOR $J_1$ , $J_2$ and $G$ CALCULATION

After a point by point coordinate transformation (180), the crack front is straight. Let us consider the segment of crack front and the volume around this segment inside a *larger* cylindrical domain  $V$ .  $V$  is the volume of the larger cylinder,  $V_\varepsilon$  is the volume of small cylinder of radius  $\varepsilon$  around the crack front segment,  $A$  is the cylindrical surface of  $V$ ;  $A_\varepsilon$  is the cylindrical surface of  $V_\varepsilon$  and  $A_1, A_2$  are side surfaces of  $V$ . Note

that at any differential segment along the crack-front, the considered domain  $V$  is still much smaller than the overall dimensions of the structure.

Then, in general, we can redefine the near-tip parameters  $J_k$  and  $G_{III}$  as:

$$J_k f = - \int_{A-A_\epsilon} \left( W n_k - \sigma_{ij} \frac{\partial u_i}{\partial x_k} n_j \right) s dA \quad (184)$$

$$G_{III} f = - \int_{A-A_\epsilon} \left( W^{III} n_1 - \sigma_{3j} \frac{\partial u_3}{\partial x_1} n_j \right) s dA \quad (185)$$

Here,  $s = s(x_1, x_2, x_3)$  is an arbitrary but continuous function which is equal to zero on  $A$ ; and nonzero (equal to 1) on  $A_\epsilon$ ; and  $f$  is the area under the  $s$ -function curve along the segment of the crack-front under consideration.

Using the divergence theorem, we have the following representation of  $J_k$ :

$$\begin{aligned} J_k f = & - \int_{V-V_\epsilon} \left( W \frac{\partial s}{\partial x_k} - \sigma_{ij} \frac{\partial u_i}{\partial x_k} \frac{\partial s}{\partial x_j} \right) dV - \int_{V-V_\epsilon} \left[ \frac{\partial W}{\partial x_k} - \frac{\partial}{\partial x_j} \left( \sigma_{ij} \frac{\partial u_i}{\partial x_k} \right) \right] s dV \\ & + \int_{A_1+A_2} \left( W n_k - \sigma_{ij} \frac{\partial u_i}{\partial x_k} n_j \right) s dA \quad k = 1, 2 \end{aligned} \quad (186)$$

This expression represents a further variant of the virtual crack extension method, but the elimination of the actual process of virtual crack extension during the development of (186) allows us to use any  $s$ -function for the calculation of  $J_k$ . Thus, we have a new and computationally more appealing interpretation of the VCE approach.

In the case of the presence of nonelastic (thermal and plastic) deformations we can define  $W$  as

$$W = \int \sigma_{ij} d\epsilon_{ij} \quad (187)$$

$$\epsilon_{ij} = \epsilon_{ij}^e + \epsilon_{ij}^p + \epsilon_{ij}^t \quad (188)$$

where  $\epsilon_{ij}^e$ ,  $\epsilon_{ij}^p$ , and  $\epsilon_{ij}^t$  are elastic, plastic, and thermal parts of strains. Assuming that the stresses have an elastic potential, i.e.,

$$\sigma_{ij} = \frac{\partial W^e}{\partial \epsilon_{ij}^e} \quad (189)$$

the second term of (186) can have the form

$$\begin{aligned} (J_k f)_2 &= - \int_{V-V_\epsilon} \left( \frac{\partial W}{\partial x_k} - \sigma_{ij} \frac{\partial \epsilon_{ij}}{\partial x_k} \right) s dV \\ &= - \int_{V-V_\epsilon} \left( \frac{\partial W^p}{\partial x_k} - \sigma_{ij} \frac{\partial \epsilon_{ij}^{pt}}{\partial x_k} \right) s dV \end{aligned} \quad (190)$$

Here we used equilibrium equations (in the absence of body forces), and introduced the definitions:

$$W^p = \int \sigma_{ij} d\epsilon_{ij}^{pt} \quad (191)$$

$$\epsilon_{ij}^{pt} = \epsilon_{ij}^p + \epsilon_{ij}^t \quad (192)$$

It is evident that in the absence of nonelastic strains the second term of (186) is equal to zero. If the  $s$  function is equal to zero on faces  $A_1$  and  $A_2$ , then the third term of (186) will be equal to zero as well.



Considering  $G_{III}$  for the linear elastic case (in the absence of body forces), we can have from equation (185),

$$G_{III}f = - \int_{V-V_\varepsilon} \left( W^{III} \frac{\partial s}{\partial x_1} - \sigma_{3j} \frac{\partial u_3}{\partial x_1} \frac{\partial s}{\partial x_j} \right) dV - \int_{V-V_\varepsilon} \sigma_{3j} \frac{\partial \varepsilon_{ij}^t}{\partial x_1} s dV + \int_{A_1+A_2} \left( W^{III} n_1 - \sigma_{3j} \frac{\partial u_3}{\partial x_1} n_j \right) s dA \quad (193)$$

The third terms of (186) and (193) can be simplified if the faces  $A_1$  and  $A_2$  are orthogonal to the crack front ( $n_1 = n_2 = 0$  on  $A_1$  and  $A_2$ ).

Again, in Eq. (193), the second term is equal to zero if  $\varepsilon_{ij}^t = 0$  and the third term is absent if  $s = 0$  on  $A_1$  and  $A_2$ . We note that the domain integral algorithms analogous to those in (186) can be developed directly for the energy-release-rate quantities  $G_I$  and  $G_{II}$  also.

It is now easy to see the advantage of the domain-integral method over the "stiffness derivative" method for the computation of the first component of  $J$ , i.e.,  $J_1$ , for *elasto-static problems*. If in Eq. (186), the function  $s$  is taken to be zero at  $(A_1 + A_2)$ ; and if body-forces are zero, in *elasto-static* problems the second and third terms on the r.h.s. in (186) vanish identically, and  $J_1$  can be written as:

$$J_1 = - \lim_{\varepsilon \rightarrow 0} \int_{V-V_\varepsilon} \left[ W \frac{\partial s}{\partial x_1} - \sigma_{ij} \frac{\partial u_i}{\partial x_1} \frac{\partial s}{\partial x_j} \right] dV \quad (194)$$

We compare Eq. (165) of the stiffness derivative method to the above Eq. (194). Without loss of generality, we assume that the domain  $V_o$  of Eq. (165) to be the same as the domain  $(V - V_\varepsilon)$  of (194). In (165), it is clear that the integral is quadratic in  $\mathbf{q}_o$ , say of the form  $\mathbf{q}_o \cdot \mathbf{A}_o \cdot \mathbf{q}_o$ . Thus the domain integral method gives *directly* the matrix  $\mathbf{A}_o$  [which is equivalent to  $d\mathbf{K}_o/da$ ] without using the finite difference method to evaluate  $(d\mathbf{K}_o/da)$  as in (165) of the virtual crack extension or the stiffness derivative method.

### 7.3 CHOICE OF $s$ -FUNCTION

It is natural to use a parametric representation of function  $s$  inside any element as:

$$s = N^I s^I \quad (I = 1 \dots 20) \quad (195)$$

where  $N^I = N^I(\xi, \eta, \zeta)$  — quadratic shape functions,  $I$  is the node number. We suppose summation over repeated indices. Then the  $s$ -function should be defined in terms of (195) by using 1 or 2 elements in  $x_3$  direction for the crack front disk. Usually it is not useful to have a  $s$ -function more complicated than a linear function in the radial direction. Several simple functions  $s$  are discussed below:

(a) The disk has two elements in  $x_1$  and in  $x_3$  directions respectively. The function  $s$  can be defined on the small tube of radius  $\varepsilon$ , for both the elements along  $x_3$  (with  $\zeta$  being the natural coordinate along the crack front segment), as:

$$\begin{aligned} \text{El. 1} : \quad s &= \frac{1}{2}(1 + \zeta) \\ \text{El. 2} : \quad s &= \frac{1}{2}(1 - \zeta) \end{aligned}$$

The area under the  $s$ -function curve along the meridian of the surface of the small tube or on the crack-front ( $\varepsilon = 0$ ) is equal to

$$f = \frac{1}{2}(\Delta_1 + \Delta_2)$$

(b)

$$\begin{aligned}
\text{El. 1} &: s = \frac{1}{2}(\zeta^2 + \zeta) \\
\text{El. 2} &: s = \frac{1}{2}(\zeta^2 - \zeta) \\
f &= \frac{1}{3} + (\Delta_1 + \Delta_2)
\end{aligned}$$

(c)

$$\begin{aligned}
s &= (1 - \zeta^2) \\
f &= \frac{2}{3}\Delta
\end{aligned}$$

(d)

$$\begin{aligned}
s &= 1 \\
f &= \Delta
\end{aligned}$$

Assuming a  $s$ -function that is linear in the  $r$ -direction we have its value for a particular point:

$$s = s_o \frac{r - r_\varepsilon}{r_f - r_\varepsilon}$$

where  $s_o$  – the value of  $s$  function at the point  $r = r_\varepsilon$ ,  $r$  is the distance of the point in question from the surface of small tube,  $r_\varepsilon$  is the radius of the small tube,  $r_f$  – the outer radius of crack front disk. In practice it is often useful to have one element in the  $r$ -direction for the disk and to employ degenerate quarter-point singular elements around the crack tip.

#### 7.4 FIRST TERM OF $J_k$

Using the parametric representation of displacements

$$u_i = N^J u_i^J \quad (196)$$

where  $i$  is the direction of crack front coordinate system and the superscript  $J$  is the node number, it is possible to have such an expression for the calculation of the first term of  $J$ -integral of Eq. (186):

$$(J_k f)_1 = - \int_{-1}^1 \int_{-1}^1 \int_{-1}^1 \left( W \frac{\partial N^L}{\partial x_k} s^L - \sigma_{ij} \frac{\partial N^M}{\partial x_k} \frac{\partial N^L}{\partial x_j} u_i^M s^L \right) \det(j) d\xi d\eta d\zeta \quad (197)$$

where  $\det(j)$  is the determinant of Jacobi matrix. An effective procedure of computing  $J_k$  with several types of  $s$ -functions consists of a separate  $2 \times 2 \times 2$  integration of the expression

$$R_k^L = - \int_{-1}^1 \int_{-1}^1 \int_{-1}^1 \left( W \frac{\partial N^L}{\partial x_k} - \sigma_{ij} \frac{\partial N^M}{\partial x_k} \frac{\partial N^L}{\partial x_j} u_i^M \right) \det(j) d\xi d\eta d\zeta \quad (198)$$

and defining a scalar product:

$$(J_k f)_1 = R_k^L s^L \quad (199)$$

## 7.5 THE SECOND TERM OF $J_k$

The main difficulty in integrating (190) arises from the fact that we know precise enough values of strain energy density, and strain, only at the  $2 \times 2 \times 2$  Gauss Integration points. A possible way of integrating the derivative of such functions is to obtain the derivative at the center of the element and to perform a one-point integration.

Consider a 20-node element in local coordinate system  $\xi_i$

$$\xi_1 = \xi, \quad \xi_2 = \eta, \quad \xi_3 = \zeta$$

Let's assume that we know that values of the function  $F$  only at the integration points as  $F^{(J)}$ . Using a parametric representation, it is possible to write

$$F^{(J)} = L^{I(J)} F^I \quad (200)$$

where  $F^I$  are unknown values of  $F$  at corner nodes  $1 \dots 8$ ,  $L^I$  are linear shape functions for corner nodes,  $L^{I(J)}$  are values of shape functions at integration points  $(J)$ .

The inversion of (200) gives

$$F^I = (L^{I(J)})^{-1} F^{(J)} \quad (201)$$

The coefficients of the extrapolation matrix  $(L^{I(J)})^{-1}$  are:

$$(L^{I(J)})^{-1} = \begin{bmatrix} A & B & C & B & B & C & D & C \\ & A & B & C & C & B & C & D \\ & & A & B & D & C & B & C \\ & & & A & C & D & C & B \\ & & & & A & B & C & B \\ \text{sym.} & & & & & A & B & C \\ & & & & & & A & B \\ & & & & & & & A \end{bmatrix} \quad (202)$$

$$A = \frac{5 + 3\sqrt{3}}{4} \quad B = -\frac{\sqrt{3} + 1}{4} \quad C = \frac{\sqrt{3} - 1}{4} \quad D = \frac{5 - 3\sqrt{3}}{4} \quad (203)$$

Now, from (202) we can calculate the derivative at the center of the element:

$$\frac{\partial F}{\partial x_k} = \frac{\partial L^I}{\partial x_k} (L^{I(J)})^{-1} F^{(J)} \quad (204)$$

where, the derivatives  $\partial L^I / \partial x_k$  should be calculated for  $\xi = \eta = \zeta = 0$ .

## 7.6 THIRD TERM OF $J_k$ OF EQ. (196)

For simplicity consider the disk with  $A_1$  and  $A_2$  being orthogonal to the crack front segment. Then

$$\begin{aligned} n_1 = n_2 = 0, \quad n_3 = 1 & \quad \text{on } A_1 \\ n_1 = n_2 = 0, \quad n_3 = -1 & \quad \text{on } A_2 \end{aligned}$$

$$(J_k f)_3 = - \int_{A_1 + A_2} \sigma_{i3} \frac{\partial u_i}{\partial x_k} n_3 s \, dA \quad (205)$$

If  $A_1 = A_2$

$$(J_k f)_3 = - \int_{A_1} \Delta \left( \sigma_{i3} \frac{\partial u_i}{\partial x_k} \right) s \, dA \quad (206)$$



where  $\Delta(F) = (F)_{A_1} - (F)_{A_2}$ .

Assuming that every function is linear in the  $x_3$  direction and using the values of functions at integration points  $\zeta = \pm(1/\sqrt{3})$ , it is possible to define  $\Delta F$  as

$$\Delta F = \sqrt{3} \left[ F \left( \zeta = \frac{1}{\sqrt{3}} \right) - F \left( \zeta = -\frac{1}{\sqrt{3}} \right) \right]$$

Then

$$(J_k f)_3 = - \int_{-1}^1 \int_{-1}^1 \sqrt{3} \Delta \left( \sigma_{i3} \frac{\partial u_i}{\partial x_k} \right)^{(T)} s \det(J) d\xi d\eta \quad (207)$$

where  $\Delta F^{(T)} = F(\zeta = \frac{1}{\sqrt{3}}) - F(\zeta = -\frac{1}{\sqrt{3}})$ .

Examples of three-dimensional  $J_1$  computations using the domain-integral methods may be found in [50, 66]. Application of the "domain-integral" type evaluation of the crack-tip integral parameters in viscoplastic dynamic crack propagation at fast speeds, has been discussed in [68].

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## Summary Remarks

### Guest Speaker Honorable Tom Lewis (R-FL)

I am pleased to have the opportunity to address the 1991 International Conference on Aging Aircraft and Structural Airworthiness.

Since the first conference, held June 1, 1988, both the level of interest and the topics covered, have grown. As many of you know, the first conference was just five weeks after the Aloha Airlines accident. At that time, an FAA research program in aging aircraft did not exist. Today, it is funded at \$13 million with a significant increase proposed for fiscal year 1993.

This is good progress, but more needs to be done. We in the aviation sector must push for more research. I say we, because I have more than 20 years experience in aviation technology, almost 40 years experience as a pilot and over 45 years as a passenger.

Over the past three years the flying public has not forgotten the Aloha accident. In fact, the level of anxiety over fear of a sudden catastrophic failure has increased. A one in 10 billion occurrence, which was the description used to describe the Aloha accident, does not reassure the flying public.

NTSB aviation safety statistics for 1989 indicate an almost 25 percent improvement over the 1979 accident rate. Over the same time period a Harris poll asked "is flying safer today than 10 years ago?" In 1979 42 percent said flying was safer. But in 1989 only eight percent said flying was safer—a five-fold decrease.

Why is the flying public more nervous about flying even though the safety record is constantly improving?

In my opinion people are afraid that they may be on the "flying safety experiment" that could have a catastrophic failure unlike any that has occurred in the history of aviation. While there is support for research programs such as FAA's aging aircraft program, there is concern that this is "tombstone technology" because it was started only after the accident.

One other well known example was the 1989 Sioux City crash of a DC-10. The United flight was over Iowa when the titanium fan disk broke apart and cut all three of the hydraulic lines that are involved in aircraft control. The crash, which killed 112 of the 296 on board, was televised and shown on the evening news. A crack, which most likely began 18 years earlier, was the cause of the engine breaking apart.

Again, this "one in a billion occurrence," is not reassuring to the flying public. People want something done to prevent the accident from happening in the first place, because this was a preventable accident.

In order to correct these safety problems, I introduced legislation in 1988 and 1990; both became public law. The first, P. L. 100-591, mandated the FAA to conduct long-term research on safety problems including aging aircraft.

The second, P. L. 101-508, mandated research to find any flaws, structural weaknesses or any other hazardous condition that could lead to catastrophic accidents and to determine technologies to correct the problem.

Specifically, P. L. 100-591 mandates that "the Administrator shall undertake or supervise research to develop technologies and to conduct data analysis for predicting the effects of aircraft design, maintenance, testing, wear, and fatigue on the life of aircraft and on air safety ..."

Initially there was resistance by the Federal Aviation Administration and many in the aviation industry to a research program to detect and correct problems before they become accidents.

The Agency has started moving toward implementing a basic research program in aging aircraft. Since 1988, four scientists, two with PhD degrees, have been hired with job descriptions in aging aircraft. When coupled with the 1990 law giving FAA authority to enter into research grants arrangements with academia, FAA can develop and manage a solid aging aircraft research program.

Another factor in developing a strong aging aircraft research program at FAA is support by the aviation community, especially those of you who have attended this conference.

Conducting long-term, high risk research is an appropriate role for the Federal government. The products of this research could benefit the aviation industry and the flying public.

Since the first aging aircraft conference, many task forces have been formed and made recommendations. These task forces are largely directed at correcting today's problems. Our legislation addresses the aging aircraft problems of tomorrow with a goal of accident prevention.

Finally, a few words on how we pay for this mandated research. The Aviation Trust fund, which has a surplus of over \$15 billion, the last time I looked, is the source of research funds. If we just take the interest in treasury notes, at 8 3/4 yield, it will generate \$324 million annually in interest. This would amount to a 50 percent increase in current R&D funding.

Maybe we need legislation to establish an endowed research program from the Trust Fund surplus.

But, I don't need to tell you about the Trust fund problems. However, it does irritate me that we have a huge surplus and the FAA R&D and aging aircraft programs are struggling with meager funding.

I hope I can count on you to work with me to strengthen FAA's aging aircraft program.

Thank you.

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